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ELECTROMAGNETIC COMPATIBILITY VERIFICATION FOR THE CENTAUR AND SURVEYOR SPACE VEHICLES

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ABSTRACT

A series of five instrumented electromagnetic compatibility tests were performed on interface circuits between the Surveyor spacecraft and Atlas-Centaur launch vehicle. These tests were performed to verify that the spacecraft would not be susceptible to conductively induced effects from the launch vehicle and complex electromagnetic environment. Initial tests employed spacecraft and launch vehicle simulators in conjunction with their complementary vehicle. The final test was performed on completely assembled flight configuration equipment. Test results indicated the spacecraft, launch vehicle, and launch complex were compatible and that conducted interference levels were below specified limits. Numerical data are presented.

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SUMMARY

This report describes a program of planning, testing, and corrective action to assure electromagnetic compatibility for the space launch vehicle and payload of a major NASA project. The space booster involved was the Atlas-Centaur first- and second-stage combination, and the payload was the Surveyor spacecraft. The approach employed was to consider the total assembly as a collection of systems, subject to internal interfacing problems as well as externally generated problems arising from the launch complex. The interface of particular concern was that between Centaur and Surveyor, and Surveyor and ground support equipment (GSE). Atlas-Centaur-GSE compatibility had already been demonstrated in the launch vehicle research and development phase of the program.

Test planning and execution included five major steps, with sufficient intervening time to allow for corrective action, if necessary, before the next step. The goal was to arrive at a launch configuration for which no electromagnetic compatibility problems were expected, so that the last test would confirm overall success rather than provide information for diagnosis and further corrective action. The test program was planned to extend from May 1965 to March 1966. Favorable results generated high confidence in actual Atlas-Centaur-Surveyor-GSE electromagnetic compatibility. All seven spacecraft in the program were successfully launched.

INTRODUCTION

Centaur is the allegorical title chosen for the upper stage of the two-stage Atlas-Centaur intermediate-size launch vehicle. The Centaur was the first U. S. rocket to use liquid hydrogen fuel successfully and to restart hydrogen engines in space. Centaur's first missions were to place the Surveyor spacecraft into a lunar intercept trajectory, with an accuracy compatible with Surveyor's midcourse correction capability.

Surveyor designates a spacecraft designed for lunar soft landings. It consisted of solid and liquid fueled rockets, an integral tripod landing structure, solar cell panels, omidirectional and high gain antennas, and various electronic and scientific equipment. Surveyor's mission was to execute a soft landing on the moon in order to obtain television pictures and perform other experiments to determine the physical characteristics of the lunar surface to support the Manned Lunar Landing Program.

It was essential that the Atlas-Centaur-Surveyor and associated ground equipment be compatible in many ways, but the subject of this report is electromagnetic compatibility (EMC).

EMC is the abbreviation used to describe the harmonious coexistence of sensitive electrical circuits with those requiring larger amounts of electric power. Electromagnetic interference (EMI) is the converse. EMI involves conducted, induced, and radiated effects, which are all frequency dependent. The purpose of this investigation was to examine conducted and induced interference only. Radiated interference was checked during equipment qualification tests and during major prelaunch tests. Since the frequency spectrums and power levels of all airborne and ground radiators were known, flight equipment was subjected to this type of radiated interference during qualification testing. During major prelaunch tests, radiated EMC was checked by exercising all electrical systems, both airborne and ground, in various combinations and noting interactions, if any. Suitable instrumentation was employed to detect and identify conducted and induced interference at the booster-spacecraft electrical interface.

Historically, the term "EMC" has also been adopted to designate "Electromagnetic Coordination" - stemming from the days when telephone and power utilities cooperated in resolving their EMI problems. (In the present case, EMC, in both senses, was achieved between the systems and between representatives of the various members of a technical task force set up for this purpose.)

The NASA manual, "Electromagnetic Compatibility Principles and Practices" (ref. 1), describes the idealized situation (hypothetical Project Achilles) in which EMC design efforts are accomplished in the early phase of the program. In actual practice however, booster and spacecraft designs often proceed independently, and mutual interference problems are not faced until the systems are jointly tested - in actuality, or in simulation. This may result in an emergency remedial crash program with last-minute "quick-fixes." The Centaur-Surveyor program addressed itself to the EMC question 2 years (1964) before the first operational launch (1966). The first step in this campaign was to seek out and suppress noise sources in ground support equipment (GSE) at the launch complex and at the Combined Systems Test Site (CSTS) in San Diego.

Although the Centaur space vehicle was not designed originally as the booster for the Surveyor spacecraft, electrical equipment was designed for qualification to MIL-I-26600 and MIL-STD-826 type specifications. Similarly, the Surveyor spacecraft, although not designed specifically for EMC with Centaur, was designed and built to sound design standards of EMI suppression and susceptibility. Nevertheless, electrical interface parameters were not defined in the early stages of either project, and Centaur-generated interference characteristics were unknown. Initially, it was the position of spacecraft engineers that tolerable conducted interference levels of the spacecraft were on the order of millivolts. Booster engineers, on the other hand, believed that actual interface levels were on the order of volts. Since Centaur equipment is designed to withstand ±50-volt (peak) pulses for 500 microseconds duration, the levels of interference produced by the spacecraft at the interface were well within the capability of Centaur equipment. The efforts in this study were then primarily directed toward spacecraft tolerance levels. Contractually, the spacecraft manufacturer was not required to meet any specific interface tolerance levels. As a result, when the decision was made to mate booster and spacecraft, there was no reason to believe they were necessarily compatible, electromagnetically. So, it became a matter of some urgency to assure and verify EMC. Recent history has clearly shown that the EMC concept cannot be ignored with equanimity. The case of Ranger 6, and EMI incidents which occurred on Explorer C, Saturn SI-3, 4, 5, Bomarc, and other missiles are described in reference 1 (pp. 1-15, 3-3, and 4-21).

The purpose of this investigation then, was to remove all reasonable doubt concerning possible jeopardy of the lunar mission due to EMI. Theoretical knowledge alone, or even engineering analysis, does not materially increase assurance in mutal EMC. Only actual tests in flight configuration can really provide the confidence which management requires before committing multimillion dollar investments. Because of tight schedules, it is seldom possible to arrive at flight configuration until launch time. It was, therefore, necessary to devise meaningful tests with various degrees of simulation, yet still retain significance and validity with regard to the actual final configuration. Highly instrumented tests were required to reconcile conflicting technical opinions.

This work was performed under the technical direction of a working group consisting of members of the Centaur and Surveyor Project Mangers' staffs and contractor representatives for the booster (General Dynamics/Convair) and the spacecraft (Hughes Aircraft Corporation). Jet Propulsion Laboratory's (JPL's) Surveyor Project Control Document No. 1 (PD-1) (ref. 2) was the official instrument of agreement between the two projects. It specified tolerable levels of conducted interference to the spacecraft at the interface between the booster and spacecraft. General Dynamics/Convair (GD/C) wrote the detailed test plans, provided personnel and equipment for all tests, and published test data. The Centaur Project Office (CPO) of NASA Lewis Research Center provided the initiative, technical direction, and chairmanship of the working group. The CPO cognizant engineers also helped prepare and approve detailed test plans, generated interface electrical schematics, actively supervised the tests themselves, and reviewed and approved the final test reports. JPL and Hughes Aircraft Corporation (HAC) concurred with the test plans, witnessed the tests, and accepted the results as satisfactory.

The first step in the EMC program was for each project (spacecraft and booster) to perform measurements on its own side of the interface to determine if the levels proposed for PD-1 (ref. 2) were indeed realistic. As a result, it became apparent that spacecraft requirements could actually be eased from original estimates. In addition, HAC and JPL were able to further desensitize some of their silicon controlled rectifier switching circuits. That work will not be described in this report. The results are reflected in Revision 3 of PD-1 in which tolerable levels were revised upward. It was agreed to conduct all subsequent tests at the new levels.

The sequence of tests was coordinated and phased-in with vehicle and facility availability and major test scheduling. Five highly instrumented tests were conducted from May 1965 to March 1966.

The last test configuration consisted of all the actual flight components used for the successful launch of May 30, 1966 which resulted in the first lunar soft landing. Six successful launches followed in the Surveyor program, which was concluded in 1968.

All of these tests and successful booster flights verified the EMC of the combination: Atlas, Centaur, Surveyor, and all GSE at the launch site.

Although the scope of this work was limited to the Atlas-Centaur-Surveyor (which is a medium-sized vehicle), results are representative of both larger and smaller boosters and spacecraft. This is because the electric circuits and components involved are similar - for the launch complex, as well as for the airborne systems. The results of these tests provide data points for other projects and serve to define EMC boundaries and call attention to potential problem areas which may be common to other projects.

Considerable work has been done in this field (refs. 3 to 10), but much of it is tutorial in nature. While general and valid scientific principles are elucidated, the technique is still as much art as it is science. In any event, specific cases such as the one described in this report always require specific proof and verification of systems compatibility.

The major significance of this work is not that successful missions have resulted. As far as EMC is concerned, the flights were anticlimatic. Rather, it is important that systems requiring EMC be fully exercised, and compatibility clearly and thoroughly demonstrated prior to flight. This high degree of assurance is not possible with many other system tests such as propulsion and pyrotechnics. Since this is true, all reasonable doubts may be erased by tests similar to those to be described, and essentially complete confidence acquired before consent-to-launch is given. This is a good example of a potential problem area which can be virtually eliminated prior to flight, and previously unknown or indeterminate risks greatly minimized. While EMC is only one of many requirements for mission success, it is an essential one.

TEST PLAN

The series of major system tests that were included in the EMC verification program are listed in table I.

TABLE I. - ATLAS-CENTAUR-SURVEYOR EMC TEST PLAN

Test number	Test configuration	Location	Date
1	Atlas-Centaur 7 and spacecraft prototype T-21	Combined Systems Test Site in San Diego, California	May 1965
2	Centaur simulator, spacecraft prototype T-21, and associated GSE	Launch complex 36A at Cape Kennedy, Florida	July 1965
3	Atlas-Centaur 7, spacecraft simulator, and launch complex GSE in major system test (Flight Acceptance Test)	Cape Kennedy, Florida	October 1965
4	Atlas-Centaur 7, spacecraft simulator, and launch complex GSE in major system test (Simulated Tank Test)	Cape Kennedy, Florida	November 1965
5	Atlas-Centaur 10, spacecraft 1, and associated GSE	Combined Systems Test Site in San Diego, California	March 1966

The special EMC tests listed in table I were conducted in addition to the normal Flight Acceptance Composite Test, Tanking Test, and Combined Readiness Test which are part of routine final prelaunch preparations.

The sequence and configuration of tests were determined by equipment availability at a given time. An actual spacecraft was not available early in the EMC program. In order to begin, however, it was agreed by the technical task group to take advantage of an available spacecraft prototype (T-21) in a composite test with Atlas-Centaur 7 (AC-7) at the CSTS in San Diego.

The T-21 was a Hughes Surveyor prototypical test vehicle (comparable to a design proof test article). It was representative of Surveyor Model A-21 flight spacecraft, series SC-1 to SC-4. It was mechanically as well as electrically identical to an actual Surveyor. Electrically, it radiated radio signals of proper frequencies and power levels, responded to commands, and provided monitoring circuits to the blockhouse and inputs to Centaur telemetry.

The initial test (identified as test 1) was run on May 22, 1965 and was intended to verify the levels outlined in table II. The circuits listed in table II were the only ones

TABLE II. - PROPOSED MAXIMUM ALLOWABLE CONDUCTED EMI FROM

ATLAS-CENTAUR-GSE ENVIRONMENT TO SURVEYOR

Signal	Related components	Maximum EMI levels, mV (peak)
Accelerometer output signal	Surveyor accelerometer amplifier	20
Analog-to-digital converter output signal	Surveyor central signal processor	200
Centaur commands High power transmitter on Extend landing gear Extend omniantennas Preseparation arming	Centaur programmer	500
GSE power External battery charge External OCR input	GSE ground power supply	100
Helium vent	GSE helium vent pulse generator	250
Main power switch on/off	GSE safety console	2500
Retro igniter safe-and-arm command	GSE safety console	250
Gyro preheat power	GSE systems test equipment assembly	Not specified
Battery charge sensing	Surveyor battery	50
Retro squib integrity	Safe-and-arm device igniter	2500
Safe-and-arm sensing	Safe-and-arm device	5000

identified by spacecraft engineers as sensitive or critical. For the purpose of this report, other lines were ignored.

Some apparent incompatibilities were noted, but no equipment malfunctions occurred on the T-21. As a result, it was agreed that HAC would reconsider and change the proposed interface requirements (table II) to more realistic levels.

Since test 1 revealed that the levels proposed in table II were not realistic, test 2 was designed to investigate further the actual levels existing. No formal criteria were used for this test, which was exploratory in nature.

In test 2, the Centaur electrical simulator provided for:

(1) Continuity of all Surveyor launch complex 36A landline functions (from Centaur umbilicals to Centaur-Surveyor interface connector)

(2) Continuity of additional Centaur-Surveyor interface landline functions:
   (a) TV light control
   (b) Air conditioning thermal sensor circuits
(3) Generation of command signals for the spacecraft (simulating Centaur flight programmer-originated commands) as follows:
   (a) Preseparation arm
   (b) Extend landing gear
   (c) Extend omniantennas
   (d) Switch to high-power transmitter
(4) Electrical termination (simulating Centaur telepack loading) and monitor test points for the Surveyor-originated telemetry and transducer interface circuits:
   (a) Surveyor analog-to-digital converter output
   (b) Surveyor accelerometers outputs

Having satisfactorily completed the first two tests, subsequent major prelaunch tests 3, 4, and 5 were designed to serve as further verification of compatibility and led to a revision of the interface document, to be described in the section Test 4. Since umbilicals were ejected during the final tests and since the tests continued in simulated flight through spacecraft separation commands, positive results were to be considered significant to enhance confidence in the EMC aspect of flight.

TEST ENVIRONMENTS

The CSTS in San Diego was designed to simulate electrically, as closely as possible, the actual launch complex at Cape Kennedy. The Combined Systems Test (CST) was the first time that Surveyor was mated to its booster. Although Atlas and Centaur were not physically mated (fig. 1), the interconnecting electrical wires were made as short as possible. Long runs of wire were included to simulate those of the launch site by appropriate reduction in gage from the umbilical tower to the blockhouse at the launch site. Except for propellant and pneumatic systems, all of the control and instrumentation equipment located in the actual blockhouse were duplicated at the CSTS. All relays and solenoids found at the actual launch complex were also duplicated and actuated at the CSTS, except for the propellant loading and pneumatic systems. Although the CSTS was not designed specifically as an EMI test bed, it has proven very valuable for this purpose. It showed, for example, in test 1 that the levels originally proposed by spacecraft engineers (table I) were not realistic and that further evaluation and verification would be required. Tests 2, 3, 4, and 5 followed (see table I).

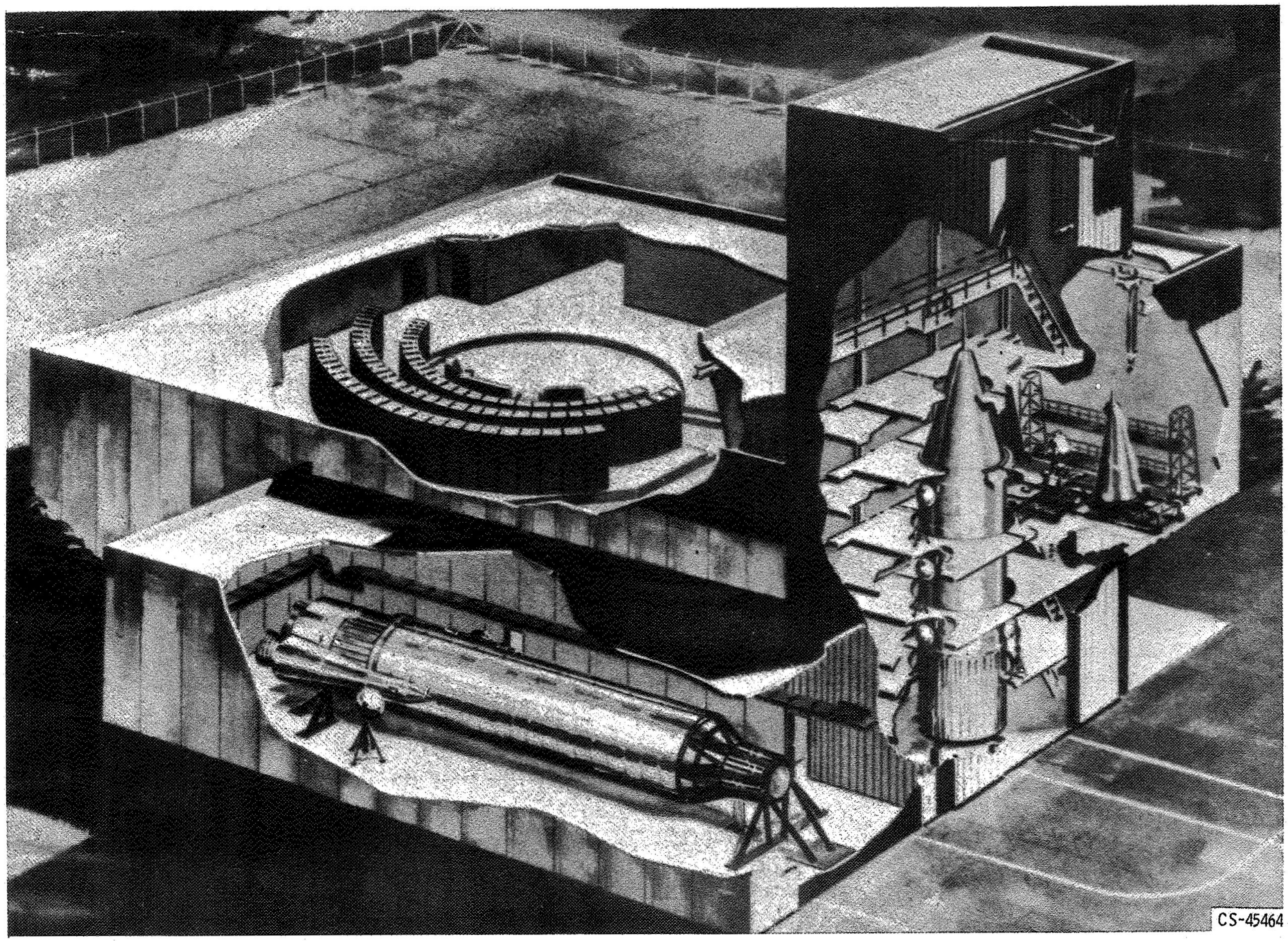


Figure 1. - Centaur spacecraft combined systems test site (located in San Diego, California).

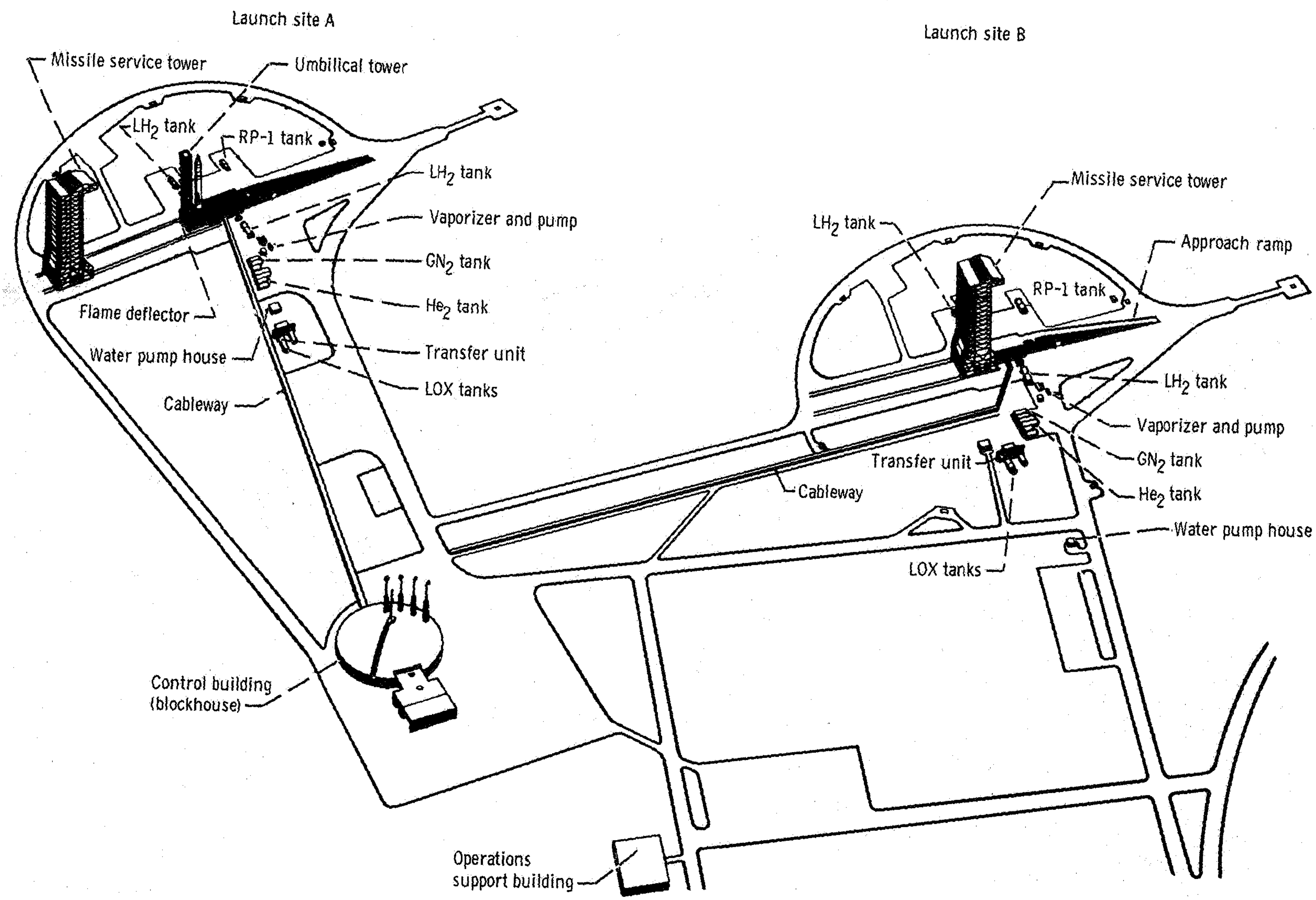


Figure 2. - Centaur launch complex 36 located at Cape Kennedy, Florida.

Figure 3. - Interior of launch complex 36 blockhouse. (Control panels shown are for launch site A.)

At Cape Kennedy launch complex 36 (fig. 2), EMI problems were anticipated because of the mutual coupling in long cable runs between the blockhouse and the vehicle (800 ft (244 m) to site A and 1500 ft (458 m) to site B) and because of the many solenoid valves (160) associated with propellant tanking. Inductive transients from these solenoids were suppressed by means of diodes in parallel with the inductive elements. In the transfer room and blockhouse (fig. 3), there are some 1200 relays. Most of these are also diode-suppressed. A classic example of transients arising from interrupting inductive circuits such as relays is shown (before suppression) in figure 4. Although this particular

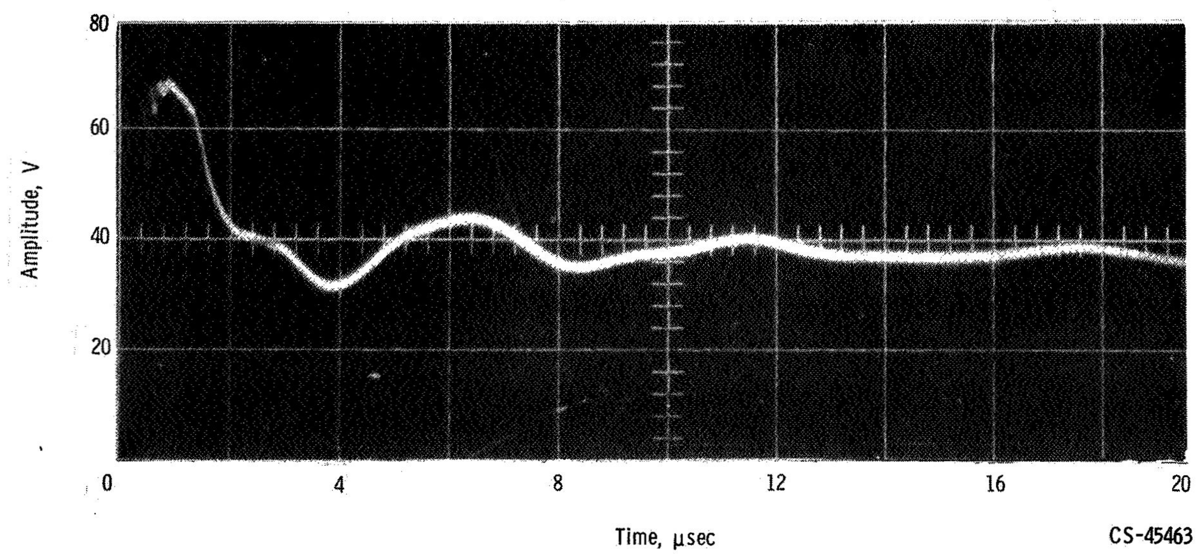


Figure 4. - Transient waveform at spacecraft simulator's latching relay.

unsuppressed transient rises only to 70 volts, peaks as high as 800 volts have been observed in other circuits when suppression is absent. This particular transient was derived from a latching relay used in the spacecraft electric simulator. The presence of this transient illustrates the importance of suppressing the test equipment itself as well as airborne and GSE sources.

BOOSTER AND SPACECRAFT CONFIGURATION

The Atlas-Centaur is classified as a medium-sized launch vehicle. With its nose-fairing, it is about 117 feet (36 m) long and 10 feet (3.1 m) in diameter (fig. 5).

The Surveyor spacecraft (fig. 6) has become well-known to the public throughout the world. For the purpose of this report, it is noteworthy that weight considerations limit

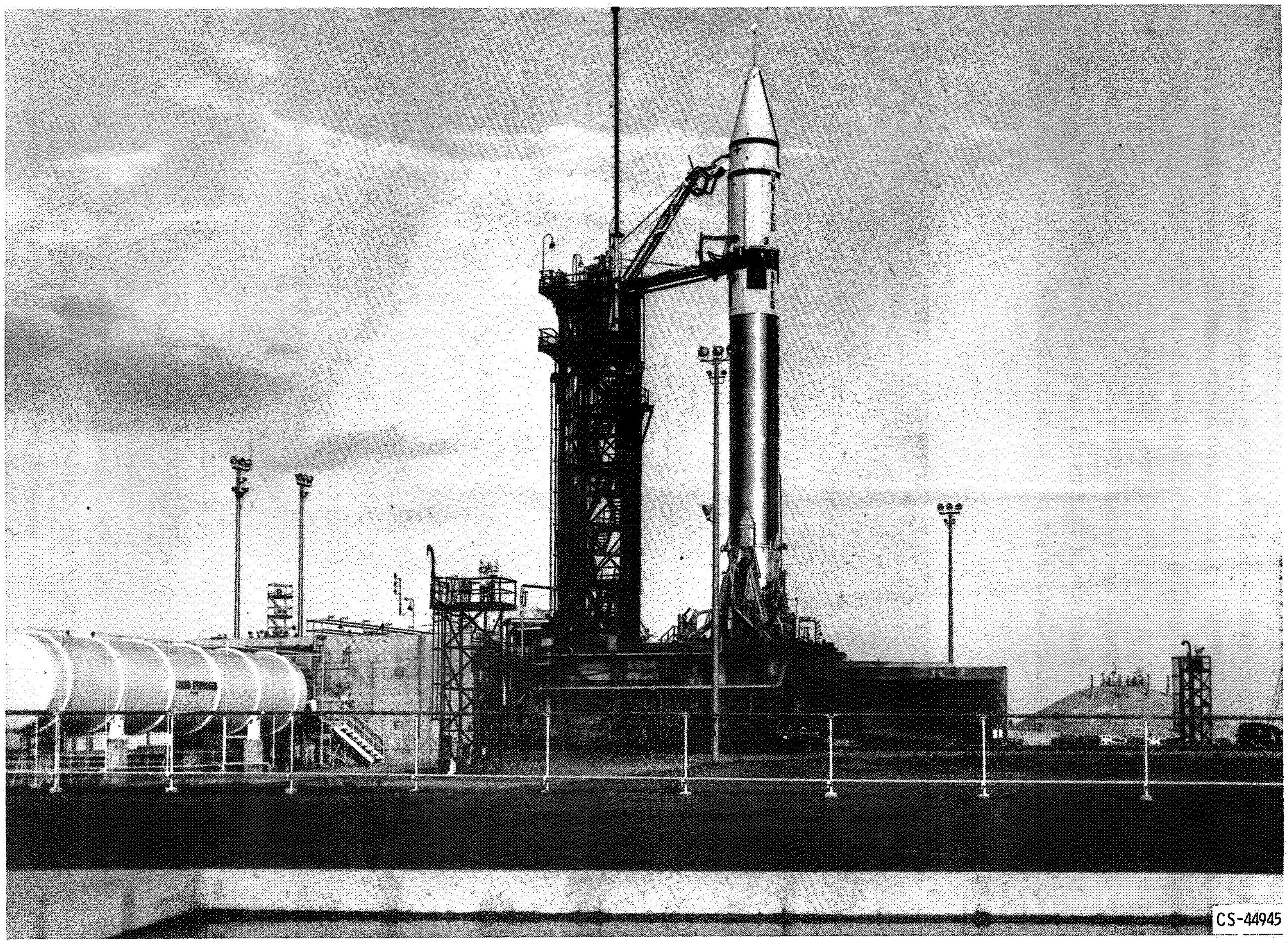


Figure 5. - Atlas-Centaur 3 ready to launch.

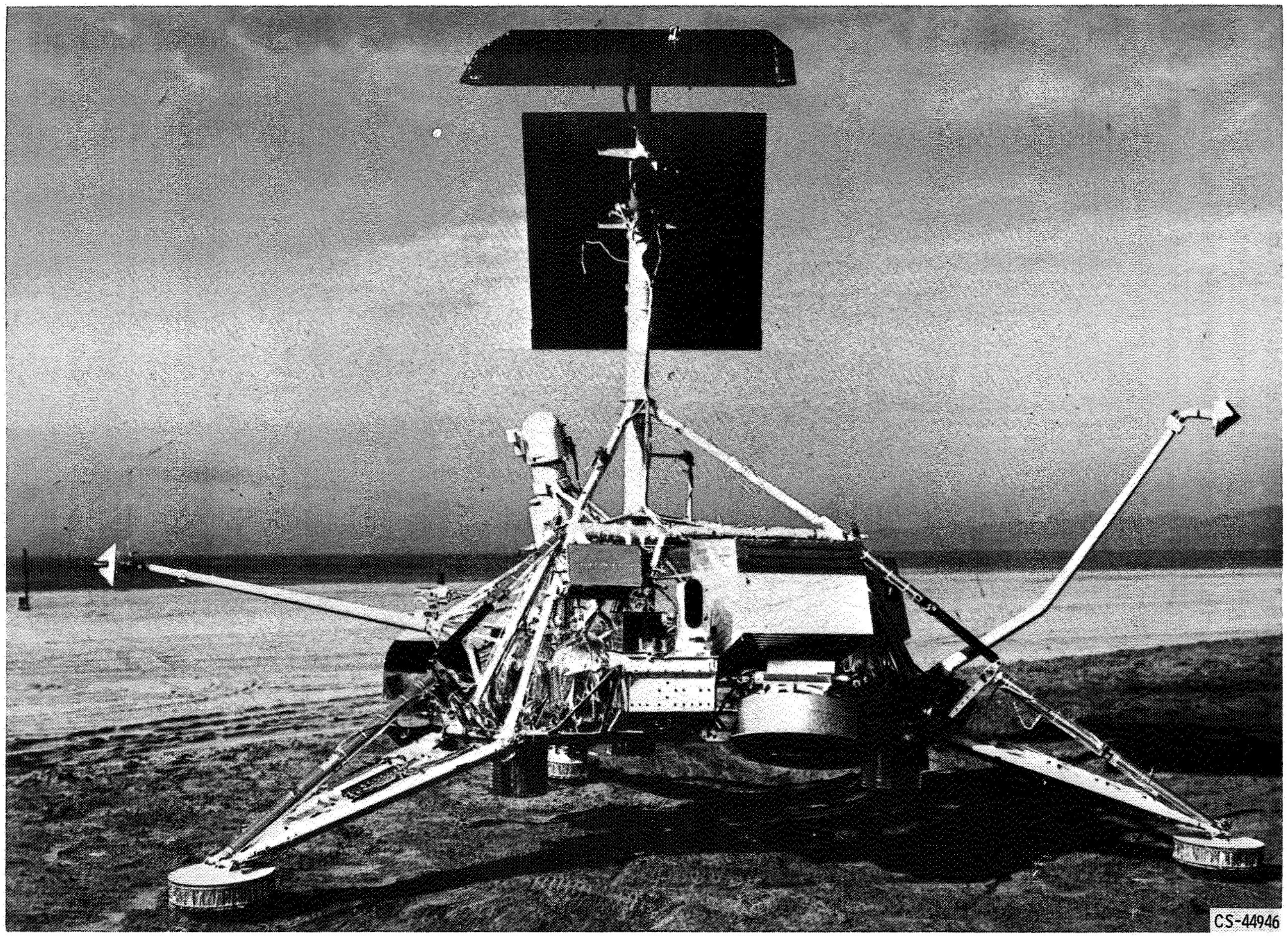


Figure 6. - Surveyor spacecraft in landing configuration.

the amount of shielding and filtering used on the spacecraft. For this reason, spacecraft circuits are generally more susceptible to EMI than circuits in which weight is not a problem. Further, since susceptibility levels were not specified by JPL, actual levels were not accurately known or defined. The levels proposed in table II were more in the nature of engineering estimates than actual hard and fast requirements.

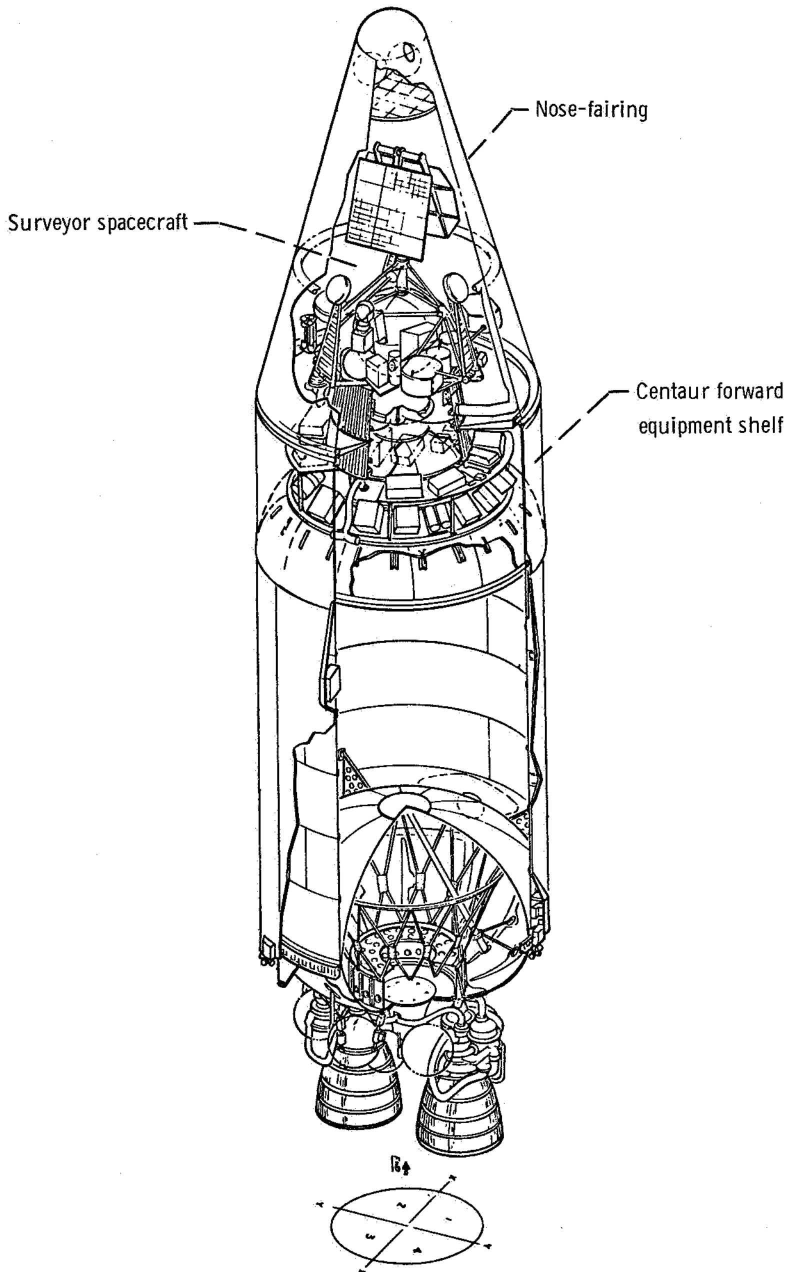


Figure 7. - Skeleton view of Centaur second stage and Surveyor spacecraft.

The spacecraft is mounted within the Centaur nose-fairing as shown in figure 7. The proximity of Centaur electric and electronic equipment on the forward shelf is clearly evident. Figures 8 and 9 show the Surveyor-Centaur separation plane and the staging disconnect. This point is not accessible during ground tests for a thermal barrier separates the spacecraft from the Centaur thermal environment. For this reason, the instrumentation used in these tests was connected at the so-called field joint connectors, located within 2 feet (0.61 m) of the separation disconnect. The field joint connectors and their relation to the Centaur forward equipment shelf is shown in figure 10.

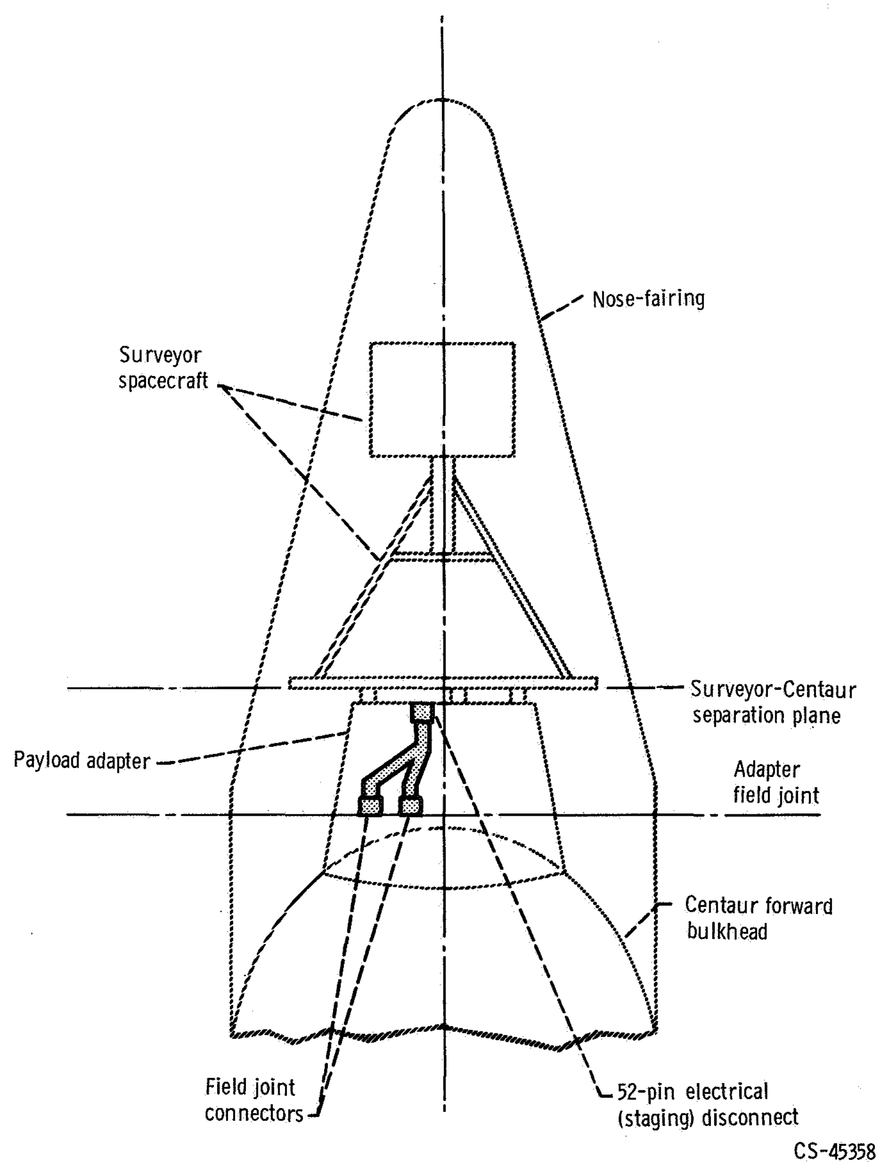


Figure 8. - Surveyor-Centaur electrical interface.

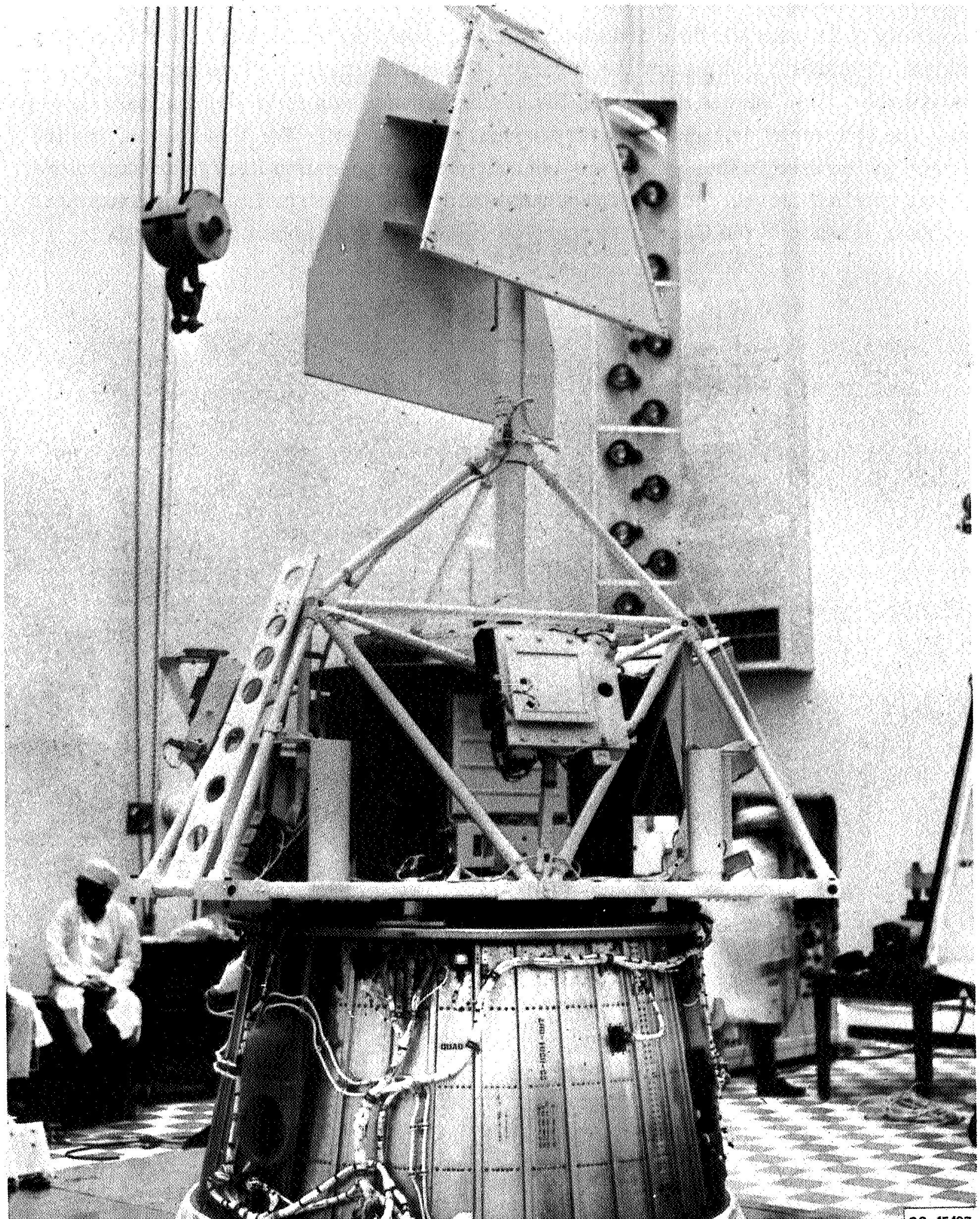


Figure 9. - Surveyor spacecraft mass model and Centaur-Surveyor adapter.

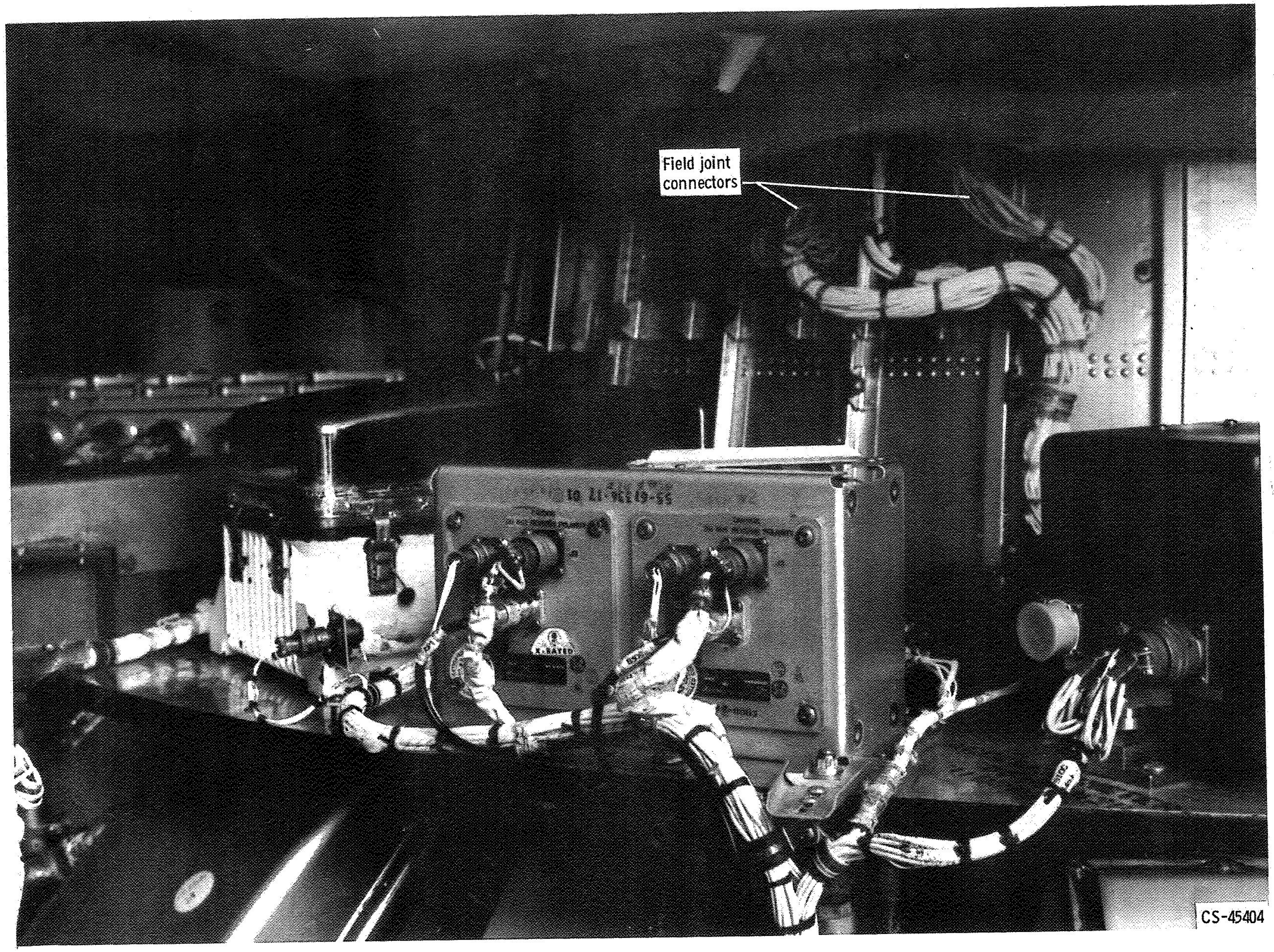


Figure 10. - Centaur equipment shelf.

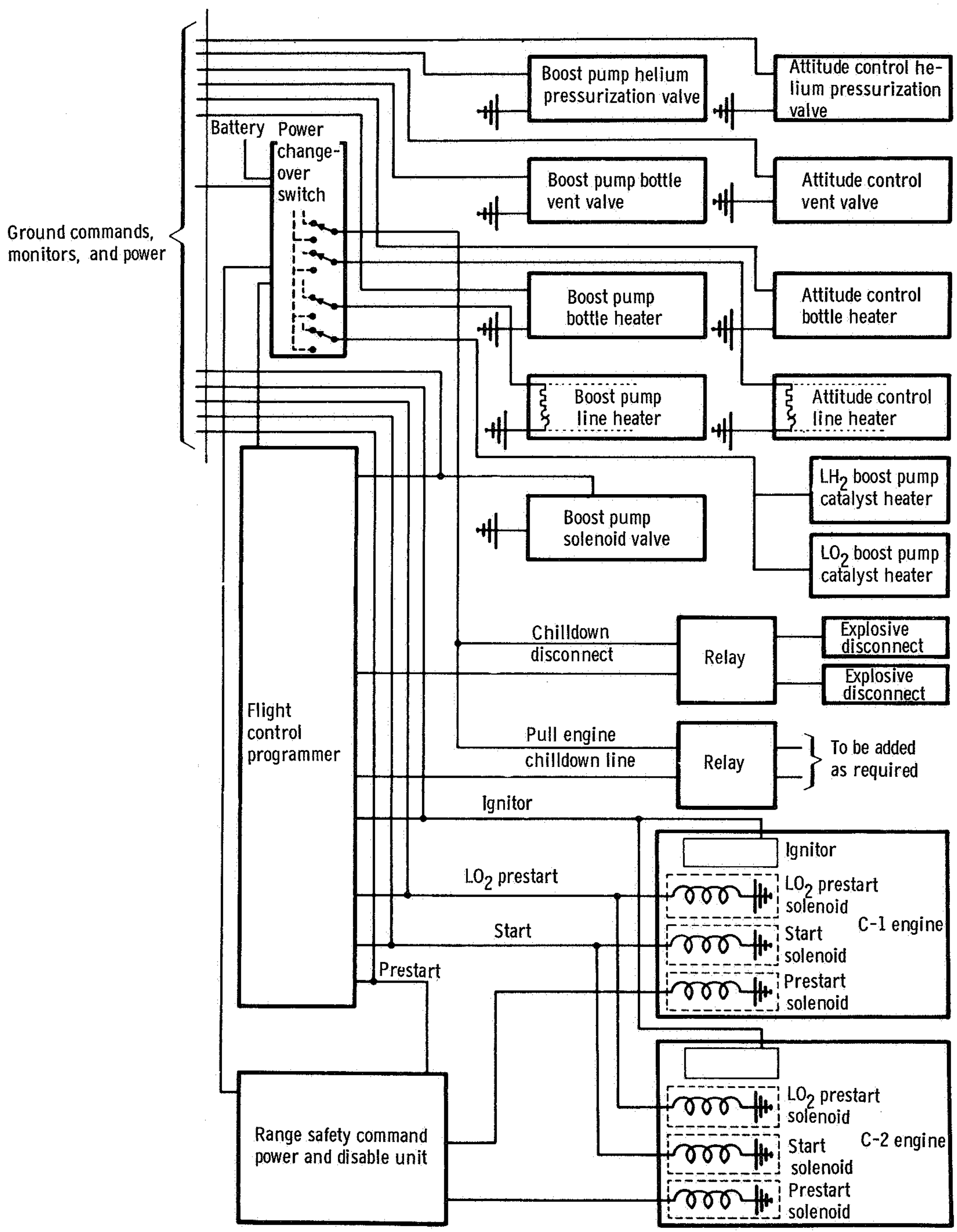


Figure 11. - Centaur propulsion electrical system.

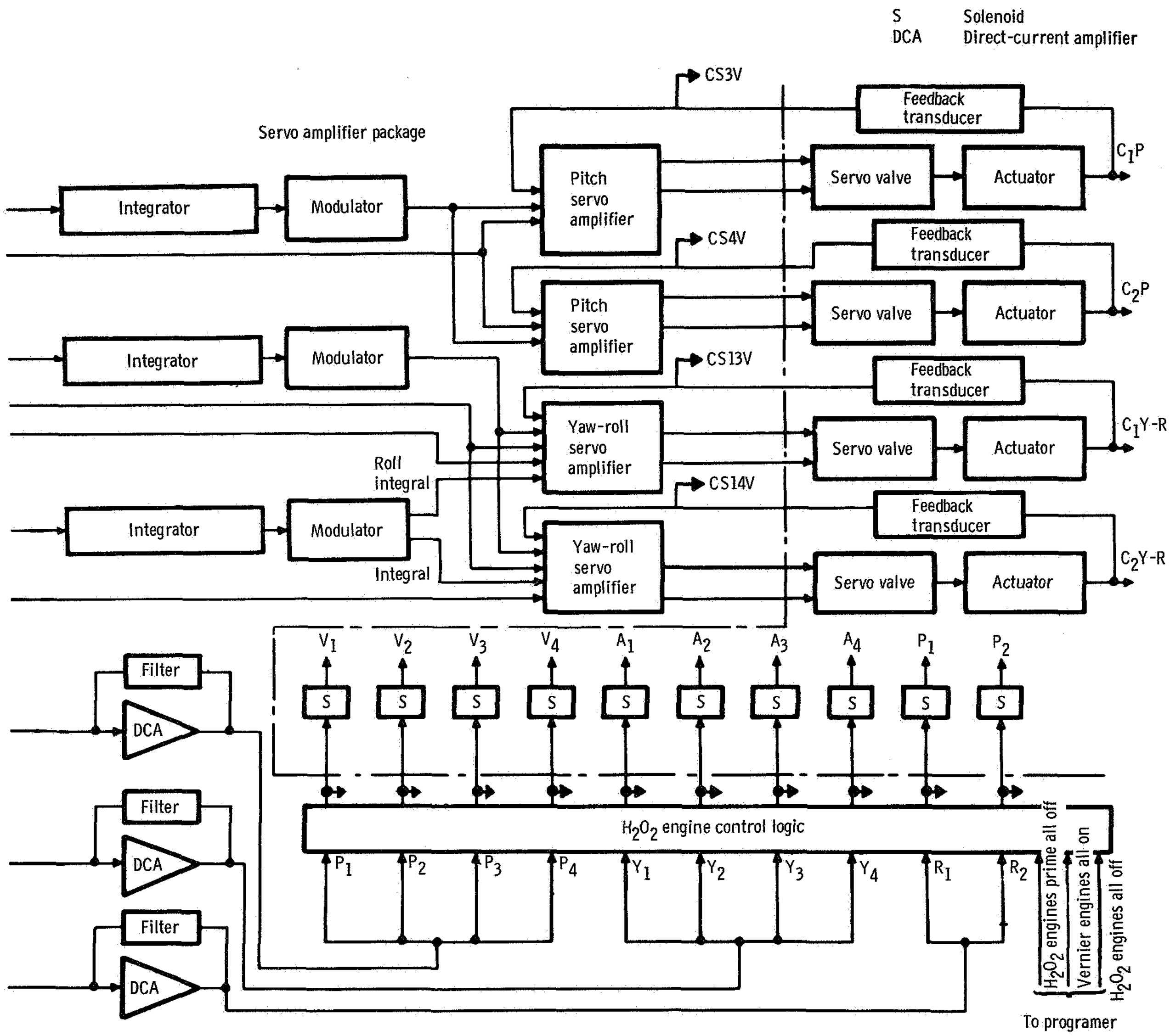


Figure 12. - Centaur flight control system.

On the Centaur side of the interface, there are numerous sources for electrical interference, especially for inductive transients. In the propulsion electrical system (fig. 11) for example, solenoids are involved with the two engines and with the boost pumps. Relays are used in connection with pyrotechnic devices. In the flight control system (fig. 12), many solenoids are used in connection with attitude control engines. Electrical transients from all of these devices should be suppressed at the source.

SPACECRAFT INTERFACE CIRCUITS, SIGNALS, AND SOURCES

The electrical interface between Centaur and Surveyor was divided into three categories:

(1) Centaur programmer commands to the spacecraft

(2) Spacecraft inputs to Centaur telemetry

(3) Control and monitoring circuits between the blockhouse and the spacecraft

Centaur and Surveyor were electrically mated by a 52-pin connector. This connector was assembled to a pyrotechnically actuated disconnect mechanism mounted on the payload adapter. The disconnect pyrotechnic was actuated by command from the Centaur programmer approximately 6 seconds prior to spacecraft separation. Figure 13 shows the details of the circuits involved.

The interface circuits included command and signal lines that interconnected with Centaur flight equipment. The rest were routed to HAC GSE via vehicle harnessing and umbilicals. At electrical disconnect, only the telemetry signal lines were energized with a few microamperes of current.

The Centaur autopilot programmer issued four consecutive 28-volt commands to the spacecraft. They consisted of an arm command 31 seconds prior to separation, followed by extend omniantennas, extend landing gear, and high power transmitter on commands, each of which was 100 milliseconds in duration.

Certain spacecraft signals were relayed and transmitted by Centaur telemetry during the launch phase of the flight. They consisted of four accelerometer signals and one analog-to-digital converter output signal. Three of the accelerometer signals originated from transducers on the retro motor attachment, while the fourth signal was generated by a spacecraft-mounted transducer. These signals are designed for 5-volt (peak-to-peak) amplitudes over a 5- to 100-hertz range. The analog-to-digital output signal consisted of 5-volt pulses at a 550-pulse-per-second repetition rate.

Ground power supply lines to Surveyor consisted of a 28-volt direct-current line for battery recharging, a 50-volt direct-current line for solar panel simulation for the optimum charge regulator (OCR) circuit, and a 27-volt direct-current line for gyro preheater power.

Blockhouse commands to the spacecraft included a main power on/off 28-volt direct-current signal, a 40-volt direct-current 30-millisecond helium vent pulse, and a retro ignitor safe-and-arm line. The main power switch consisted of a set of motor-driven contacts, while the retro safe-and-arm device placed the solid rocket motor in an armed state.

The three spacecraft monitoring signals transmitted to the blockhouse were a battery charge sense line, a retro ignitor safe-and-arm sensor, and a retro squib integrity line. The battery charge sense line was a direct input to a line voltage control circuit. The latter two signals activated indicator lamps on a console.

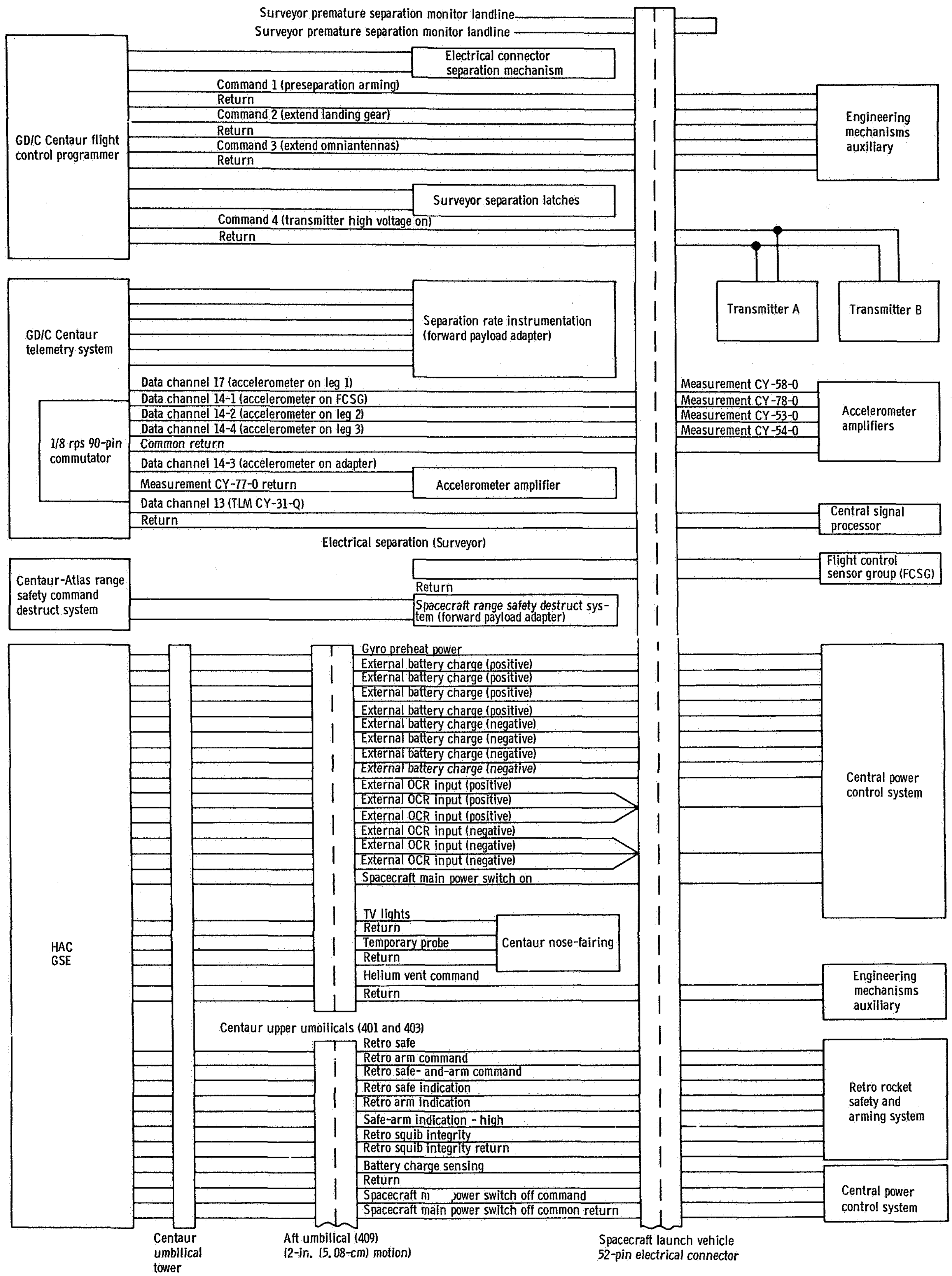


Figure 13. - Surveyor-Centaur electrical connector interface.

TEST INSTRUMENTATION

To gain access to the interface lines, a "breakout" or "sandwich" box was inserted at the field joint connectors (fig. 14). Each monitored interface line employed all the instrumentation described in this report. An absolute minimum of additional wire was introduced into the airborne circuitry, and suitable shielding and grounding techniques were employed so that the instrumentation itself would not affect normal circuit operation nor introduce foreign distrubances. Connected to the instrumentation box were a magnetic tape recorder, a recording oscillograph, cathode ray oscilloscopes, and transient detectors. The interference levels should not have exceeded one-half of the actual maximum tolerable levels specified for the spacecraft in order to be consistent with the testing philosophy of MIL-E-6051C (Electrical/Electronic System Compatibility and Interference Control Requirements). The tape recorder was limited to a 20-kilohertz response and the oscillograph to 5 kilohertz. Signal components of higher frequency were noted by means of the oscilloscope and transient detectors. The latter were unique devices designed and built by Convair and will be described in the following section. All equipment was interconnected with a common timing signal, and all test personnel were

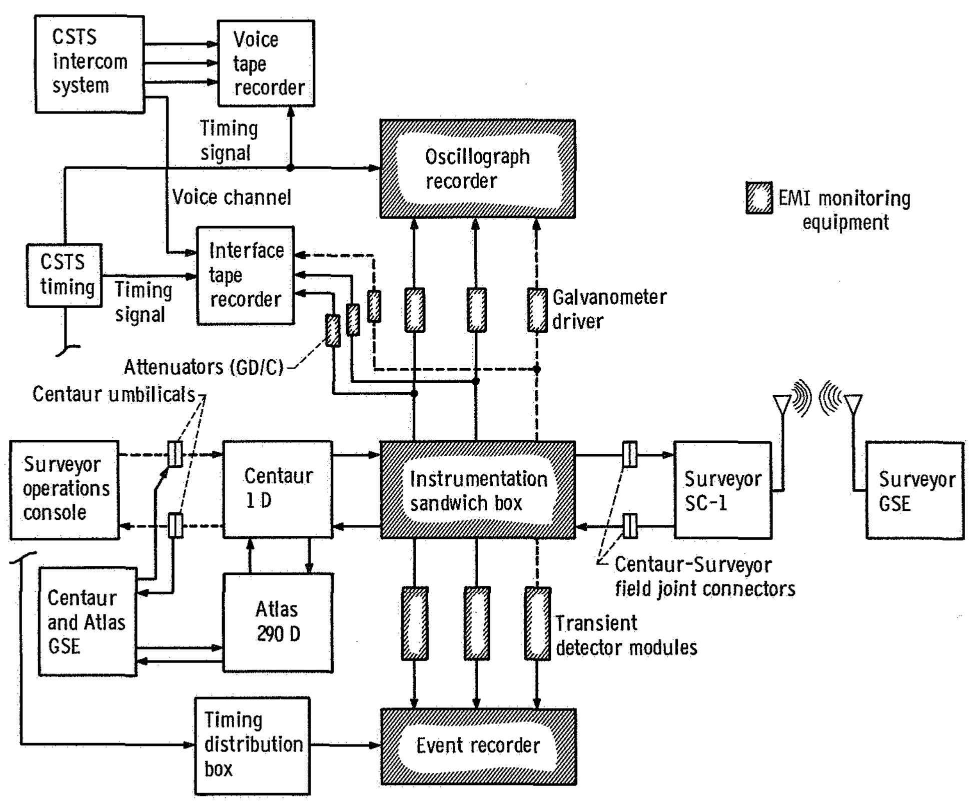


Figure 14. - Block diagram of Centaur-Surveyor interface EMI monitoring equipment for CSTS test (AC-10).

in constant voice communication. In this way, attention could be directed to imminent test events, or certain tests could be repeated at the discretion of the test conductor or at the request of any individual operator. The practice of repeating significant events or of examining the magnetic tape with the oscilloscope made it possible to use single-exposure cameras at the oscilloscope instead of continuous film recordings.

Since high-speed transient detectors were used to sense short-duration transients, tape recorders were used to define slow transients with frequency components below 20 kilohertz. The recorder was operated in the frequency modulated mode, which limited its response to 20 kilohertz. (Fourteen track capability was available.) Playback to the galvanometer oscillographs further limited the response to 5.5 kilohertz maximum. Maximum time displacement error was ±0.25 millisecond. The oscillograph and tape recorders were also used to monitor steady-state signals.

Transient Detectors

The transient detectors employed "single-shot" silicon controlled rectifiers (SCR's) as pulse switches. Triggering level of the SCR's could be accurately calibrated and reproduced. Response was obtained to pulses ranging from 1/2 microsecond to

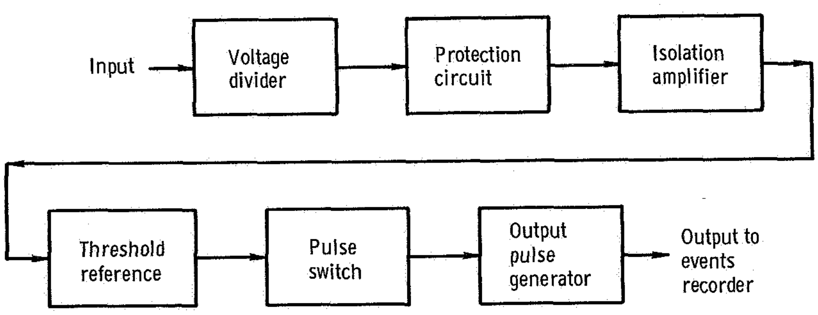


Figure 15. - Block diagram of GD/C transient detector module.

1 millisecond, in amplitude levels of 0.2 to 10, and 10 to 100 volts. The instrument was connected to respond to either positive or negative polarity. When the SCR fires, it actuates an "operations" or "events" recorder, which provides time correlation. A block diagram of the instrument is shown in figure 15. The specific nature of the transient can be examined later by repeating the event occurring at that discrete time and observing directly with the oscilloscope. Input impedance of the transient detectors was greater than 700 000 ohm, capacitively coupled.

Oscillographs

Galvanometer-type optical oscillographs were used with a frequency response of 0 to 5000 hertz. Each instrument employed six channels, and a total of two instruments were required.

Oscilloscopes

Real time monitoring was also provided by means of cathode ray oscilloscope. These were intended primarily for evaluating steady-state noise levels and are conventional laboratory instruments of 0- to 10-megahertz response capability. Photographic records were made of wave shapes of interest. Horizontal trace speeds ranged from 0.5 microsecond per centimeter to 100 milliseconds per centimeter. All test personnel were interconnected to a two-way communications network, which made it possible to call attention to specific switching events. This enabled the oscilloscope operator to anticipate transients or even request repeats of the event, if necessary.

Event Recorders

Discrete outputs from the transient detectors were monitored on chart recorders and provided correlation with switching events or other significant periods during the tests.

Spacecraft Passive Simulator Configuration

The spacecraft simulator was essentially a passive device with the exception of three relays actuating contacts to lamps indicating receipt of commands. All other circuit impedances were represented by resistances. The simulator was used in tests 3 and 4.

Because some of the interface return lines were terminated on the spacecraft frame, a potential ground loop existed between the launch vehicle and spacecraft by way of the mounting adapter. This condition was duplicated on the passive simulator by connecting 24 feet (7.3 m) of 1/0 gage wire between the Centaur forward ground plate and the common tie point within the simulator.

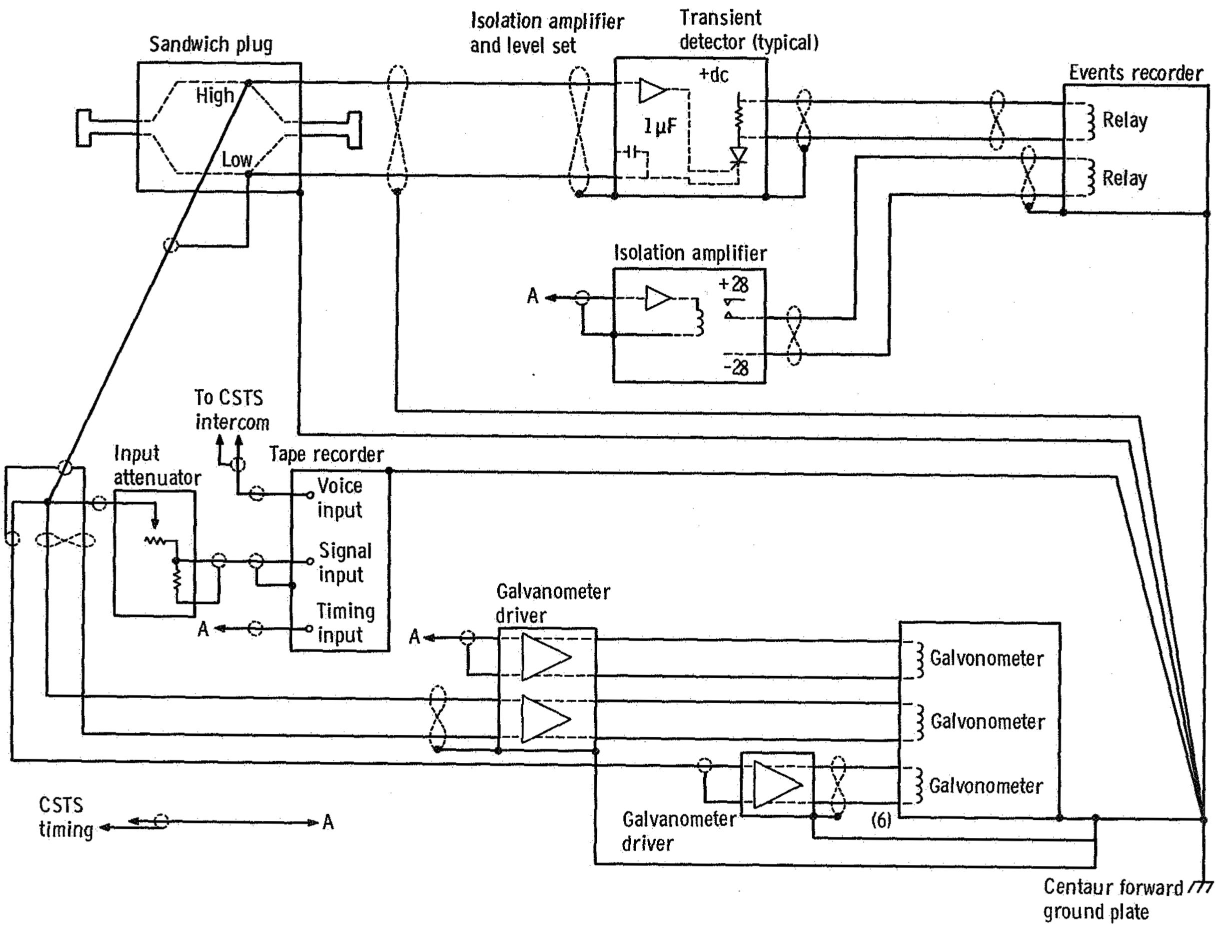


Figure 16. - Input and grounding configurations. (Isolation plugs used on all 110-volt, 60-hertz power cords.)

Instrumentation Grounding Configuration

In order to protect the integrity of the instrumentation and assure that the monitored noise levels were indicative of those on the interface lines, a single-point grounding plan was employed. Figure 16 shows the manner in which equipment chassis and wire shields were connected to the vehicle ground plate to eliminate potential interference loops.

Figure 17 shows the setup for test 1. This test employed instrumentation leads longer than desirable. Subsequent tests reduced the lead length as previously mentioned. In this test only, the T-21 spacecraft prototype was located on an upper level, while the Centaur booster was located under temporary floorboards. This first test served to alert all concerned to the apparent incompatibility then existing (May 1965) between Centaur and Surveyor. This test also provided experience for improving instrumentation on subsequent tests.

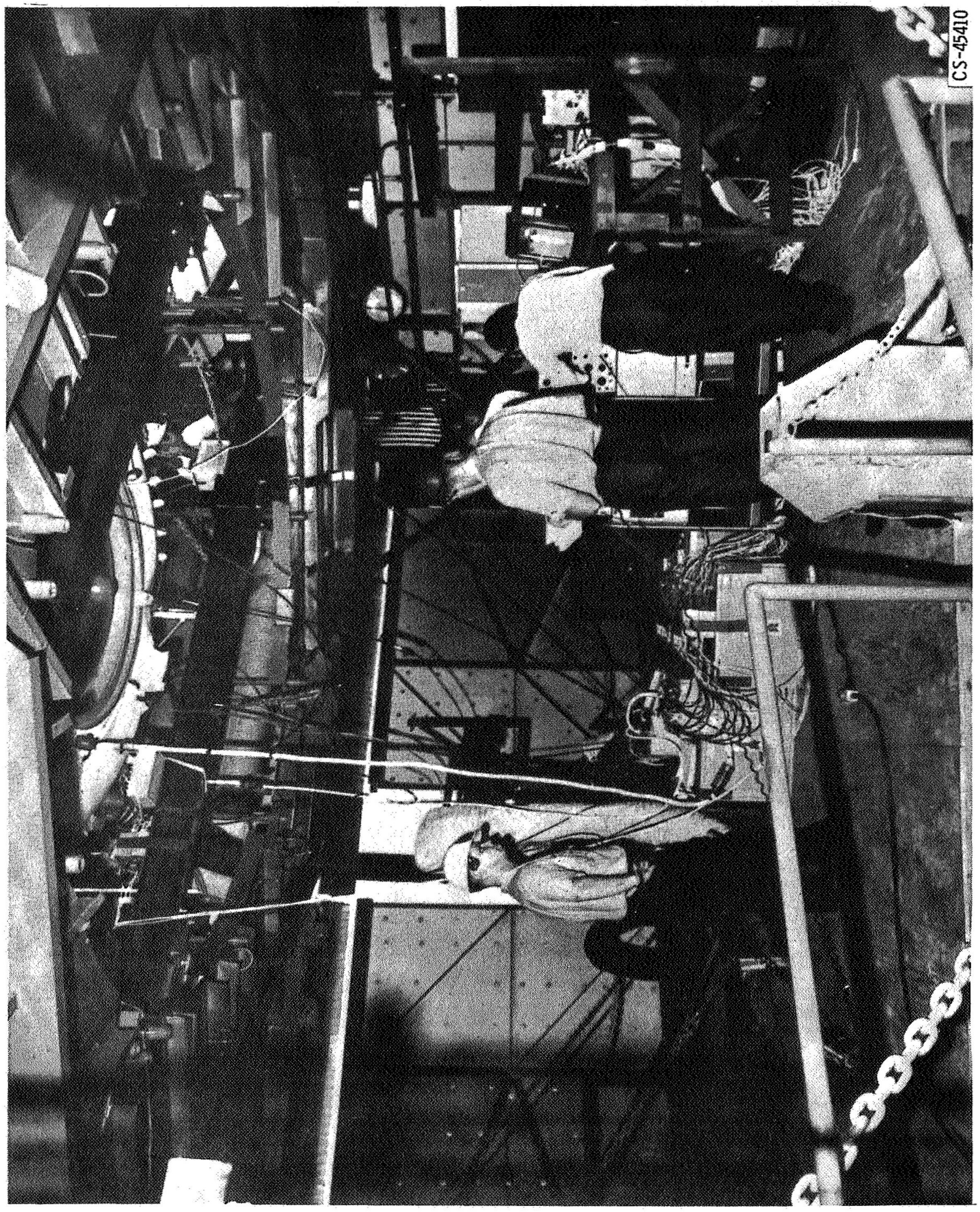


Figure 17. - Test setup 1.

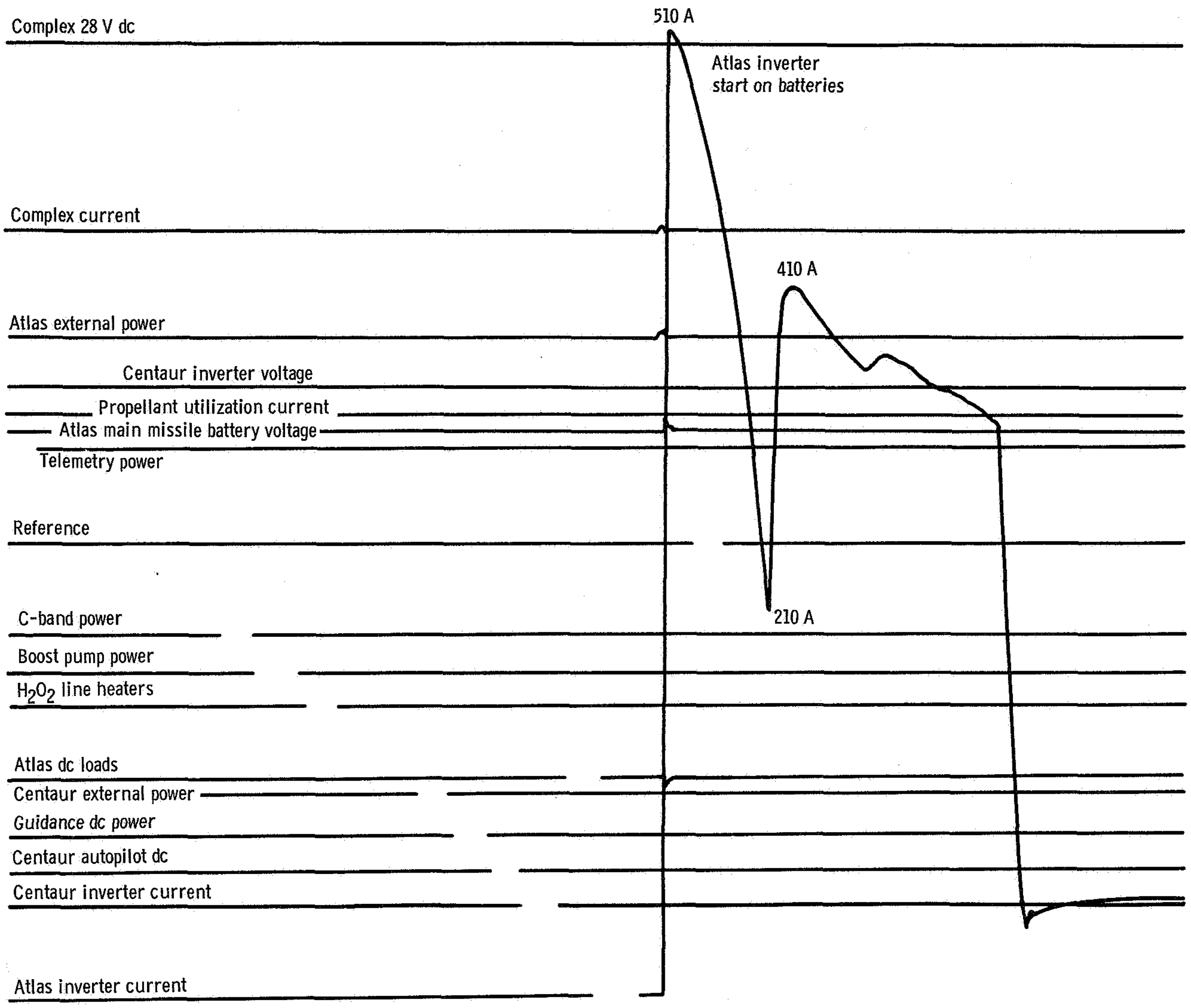


Figure 18. - Atlas direct-current traces.

As part of the normal Centaur test procedure, direct currents of all electrical subsystems are recorded, together with timing markers (fig. 18). While the resolution of this record is not sufficient to show fast transients, it is sufficient to identify and correlate switching times with possible EMI events. This figure is of particular interest because it shows a tremendous inrush of current at Atlas inverter start. In this case, it reached a peak of 510 amperes (ref. 11).

TEST RESULTS

Test 1

The Atlas-Centaur 7 and spacecraft prototype T-21 of May 22, 1965 showed a number of apparent incompatibilities between booster and spacecraft. Interface lines were monitored both on "external power" and "internal power," and only three showed continuous noise levels greater than those then acceptable to HAC and JPL as given in table II. The transient detectors were also triggered at intervals throughout the test. Most of these transients were coincident with activation of normal switching functions of the vehicle during simulated flight. The noisy lines were as follows:

Parameter (Zero-to-peak values)	Noise levels, mV		
	Proposed maximum	Actual	
		External	Internal
Accelerometer output	20	100	125
GSE power	250	280	875
Battery charge sensing	50	950	165

Spacecraft malfunctions did not occur during the test, in spite of the fact that the proposed maximum susceptibility levels (table II) were exceeded a number of times. Spacecraft commands were not falsely triggered, nor were damaging levels of conducted interference observed. In view of this, it was agreed by spacecraft engineers to reexamine the tolerance levels originally proposed by them (table II). It was further agreed by the working group to continue EMC testing at the Cape Kennedy launch complex.

Test 2

In this test, the same T-21 spacecraft prototype used in test 1 was employed. Since the Centaur stage was not available, its electrical interface was provided by a Centaur simulator described previously. The test was run on July 14, 1965 at Cape Kennedy launch complex 36A.

It was found that all noise levels were well below the external allowable tolerable levels except for OCR input line. On this line, ±40-volt (peak) transients were generated when the solar panel deployment actuator was switched on and off. In addition, 7-volt (peak-to-peak) steady-state noise at 1.3 hertz, and 19-volt (peak-to-peak) steady-state noise at 6.2 hertz were found.

Prior to this test, spacecraft engineers believed that transients on the order of ±40 volts could cause circuit damage resulting in possible mission failure, and that steady-state noise could cause errors in checkout of OCR during prelaunch operations. Subsequently, spacecraft engineers affirmed that the noise levels seen during this test were not large enough to cause either a malfunction or serious degradation of system performance. Reexamination showed that circuit damage would not occur until the voltage level reached a maximum of 80 volts. The 19-volt 6.2-hertz signal was determined to be a natural condition associated with normal OCR circuit operation.

Test 3

When the actual Atlas-Centaur vehicle AC-7 became available, another test series was begun with a spacecraft passive simulator. These tests at complex 36A were begun on October 20, 1965 in Flight Acceptance Combined Test mode of operation. Propellants were not loaded, nor was loading simulated electrically, although umbilicals were ejected at or near T - 0, which is the programmed lift-off time. Flight programmers commanded all normal flight events through simulated spacecraft separation.

The only transients noted were the expected programmed pulse commands of extend landing gear, high power transmitter on, and preseparation arming. Steady-state noise on all lines except external OCR input was well below the steady-state compatibility levels agreed to. An 8-volt (peak-to-peak) noise was observed on the external OCR input line from the start of the countdown test until the spacecraft console main power on switch was actuated, at which time the noise disappeared. The noise was attributed to inductive coupling to the line, which was essentially unterminated (open) until power was switched on. After power turn-on, the noise decreased to 0.2 volt (peak-to-peak).

Test 4

This test was actually a series of trials involving individual systems. Procedure was essentially the same as for test 3, except that propellant loading was simulated (solenoid valves were actuated). The configuration remained the same as for test 3, although individual system tests were run independently, rather than in a formal prelaunch countdown procedure. It was found that newly proposed (table III) steady-state levels were not exceeded. No undesirable transients were detected - with one exception, which was generated within the Surveyor simulator by the coil of a latching relay. This relay

TABLE III. - COMPARISONS OF EMI LEVELS

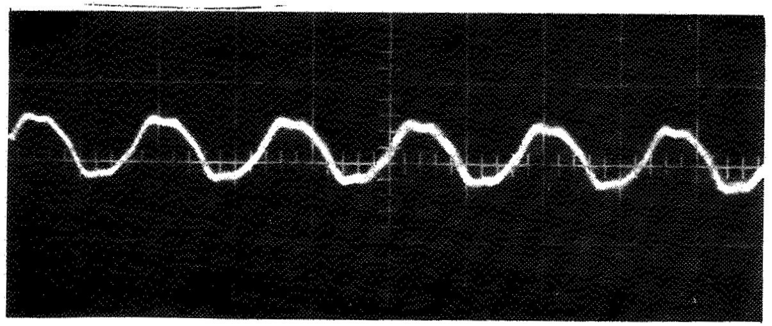
Monitored function	Maximum steady-state noise amplitude, V (peak-to-peak)					Amplitude of EMI which caused transient detector response, V				
	Test 1	Test 2	Test 3	Test 4	Test 5	Test 1	Test 2	Test 3	Test 4	Test 5
Helium vent	0.15	<5	<2	<0.01	<1	>±0.25	None	None	None	None
Extend landing gear	.4		<1			±.5				None
Extend ominantennas	.1									>50
High power transmitter on	.45									None
Preseparation arming	.25									
Battery change sensing	1.9	.4	<.2	.04	1.65	(a)				
Spacecraft accelerometer	.25	.8	<.01	.01	[b].07	(a)				
Analog-to-digital converter	.17	[c]5	<.1	.06	<.15	(a)				
GSE ground power	1.75	.3	<.2	1.6	1.86	>±.25				
External OCR input	(a)	[c]19	<.2	.4	.07	(a)	>±40			
Retro safe on	.3	<2	(a)	(a)	(a)	>±.25	None	(a)	(a)	(a)
Retro arm on	.1	<2	<1	.36	<1.25	>±.25		None	None	>-25
Main power off on	1.5	(a)	<1	.04	<1	>-2.5			None	None
Gyro preheat power	(a)	<2	<.5	.17	<1	(a)		None	>50	None
Retro squib integrity	1.25	(a)	<1	.16	<1.5	>±2.5	(a)	(a)	(a)	(a)
Safe sensing	1.0	(a)	(a)	(a)	(a)	None	(a)	(a)	(a)	(a)
Arm sensing	1.0	(a)	<1	.14	<1.25	>±5	(a)	(a)	(a)	(a)

[a]Not monitored.
[b]Normal signal.
[c]With accelerometer turned off.

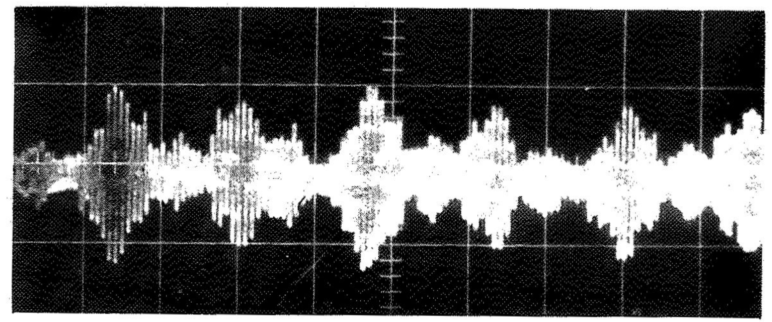
is used to simulate a motor-driven switch used in the actual spacecraft. When the 36-volt command was applied to the relay, a classic damped wave results, peaking at 68 volts (fig. 4). This is an example of large transients which occur when adequate suppression methods are not employed. This is due to the self-induced voltage L di/dt, and was never sufficiently large to affect adjacent circuits adversely. Other representative waveforms encountered are shown in figure 19.

Test 5

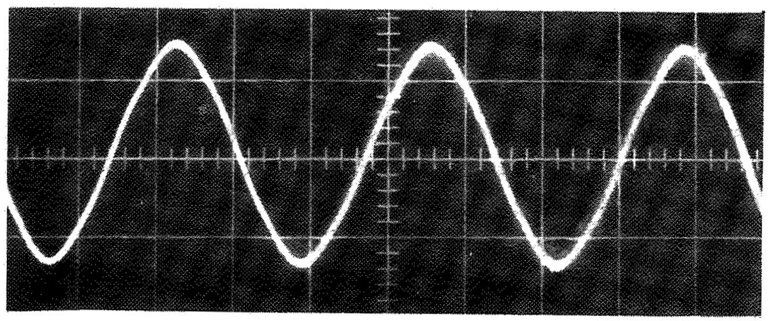
Finally, as the last in this series of tests, the actual lunar-mission configuration became available. On March 5 and 7, 1966, a Combined Acceptance Test was performed



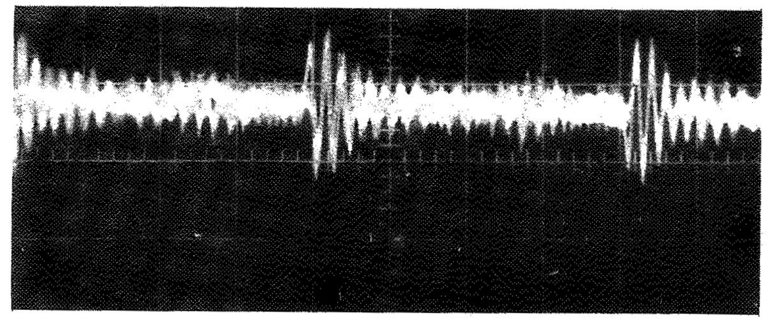
(a) Main power off/on; October 13, 1965; 2020 hours G. m. a. t.; 10 milliseconds per centimeter; 50 millivolts per centimeter (worst case).



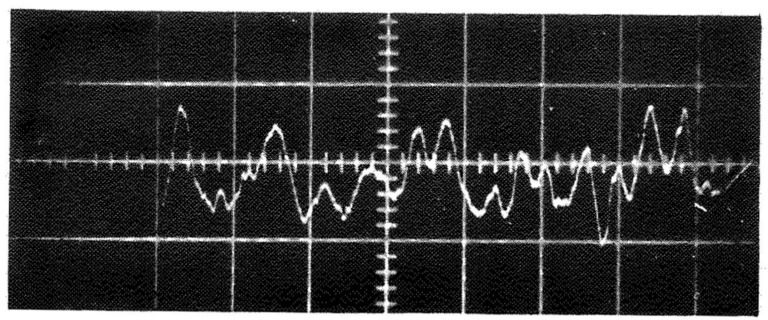
(b) Retro arm sense; October 4, 1965; 1910 hours G. m. a. t.; 5 milliseconds per centimeter; 50 millivolts per centimeter.



(c) Retro arm sense; October 6, 1965; 1505 hours G. m. a. t.; 5 milliseconds per centimeter; 50 millivolts per centimeter.

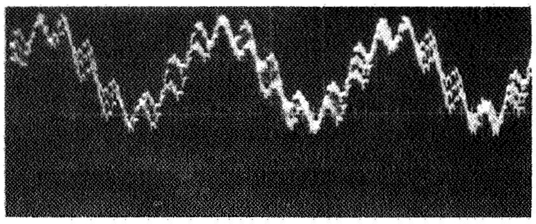


(d) Retro arm sense; October 15, 1965, 1900 hours G. m. a. t.; 2 milliseconds per centimeter; 100 millivolts per centimeter (worst case).

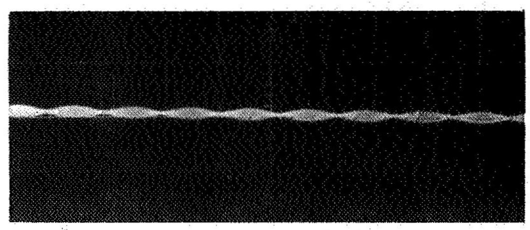


(e) Spacecraft accelerometer; March 7, 1966; 12:20 hours G. m. a. t.; 0.2 second per centimeter; 2.0 volts per centimeter.

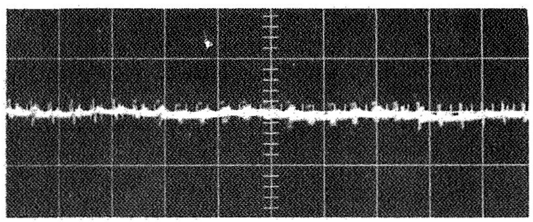
Figure 19. - Oscillograph traces on interface lines.



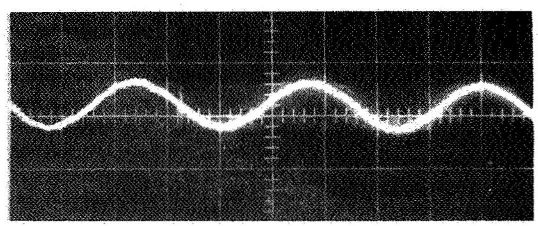
(f) GSE ground power; October 13, 1965; 1910 hours G. m. a. t.; 5 milliseconds per centimeter; 1.0 volt per centimeter.



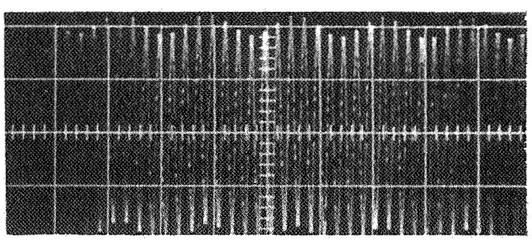
(g) GSE ground power; October 15, 1965; 1845 hours G. m. a. t.; 200 microseconds per centimeter; 50 millivolts per centimeter.



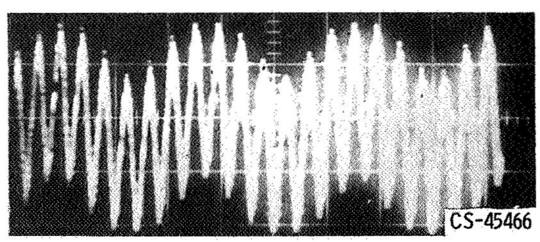
(h) External OCR input; October 1, 1965; 1855 hours G. m. a. t.; 1 millisecond per centimeter; 50 millivolts per centimeter.



(i) External OCR input; October 5, 1965; 1950 hours G. m. a. t.; 5 milliseconds per centimeter; 50 millivolts per centimeter.

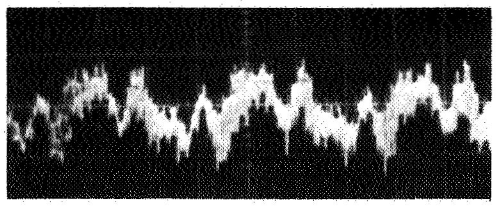


(j) External OCR input; October 8, 1965; 1535 hours G. m. a. t.; 10 milliseconds per centimeter; 100 millivolts per centimeter (worst case).

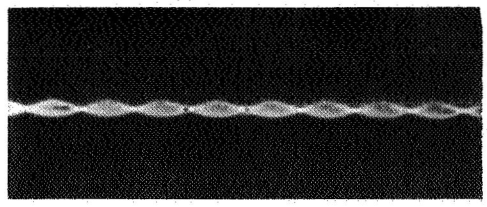


(k) Retro arm on; September 29, 1965; 5 milliseconds per centimeter; 100 millivolts per centimeter (worst case).

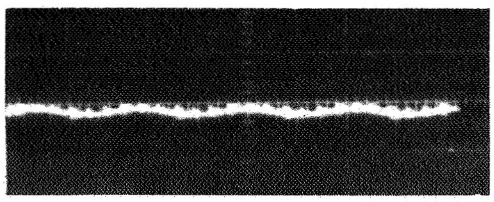
Figure 19. - Continued.



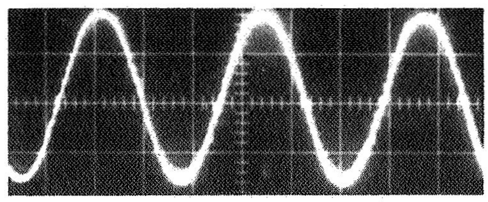
(l) Retro arm on; October 4, 1965; 1900 hours G.m.a.t.; 5 milliseconds per centimeter; 50 millivolts per centimeter.



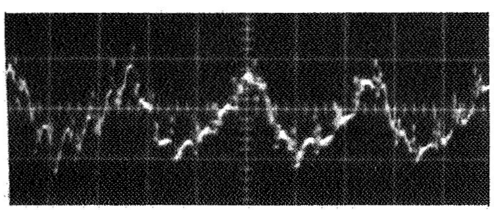
(m) Retro arm on; October 15, 1965; 1850 hours G.m.a.t.; 200 microseconds per centimeter; 50 millivolts per centimeter.



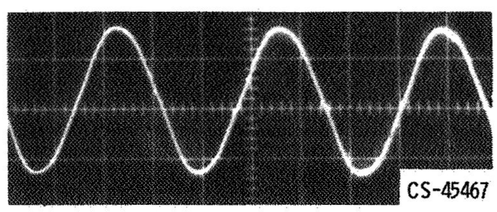
(n) Gyro preheat; September 29, 1965; 1703 hours G.m.a.t.; 1 millisecond per centimeter; 50 millivolts per centimeter.



(o) Gyro preheat; October 6, 1965; 1447 hours G.m.a.t.; 5 milliseconds per centimeter; 50 millivolts per centimeter (worst case).

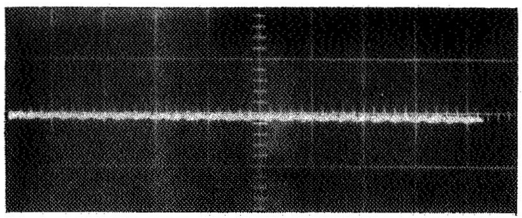


(p) Retro squib integrity; October 1, 1965; 1905 hours G.m.a.t.; 1 millisecond per centimeter; 50 millivolts per centimeter.

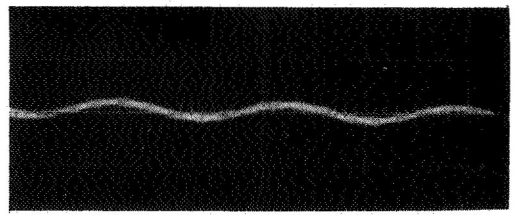


(q) Retro squib integrity; October 5, 1965; 2005 hours G.m.a.t.; 5 milliseconds per centimeter; 50 millivolts per centimeter.

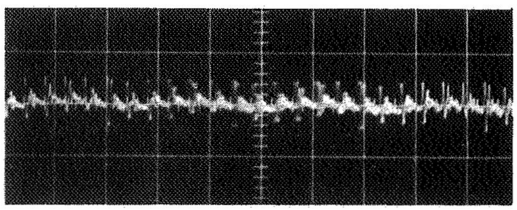
Figure 19. - Continued.



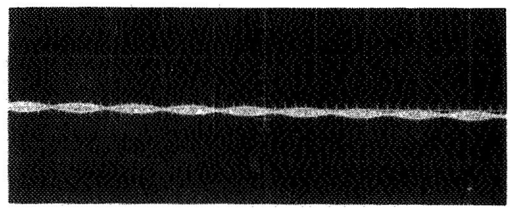
(r) Extend omniantennas; September 29, 1965; 1550 hours G. m. a. t.; 1 millisecond per centimeter; 50 millivolts per centimeter.



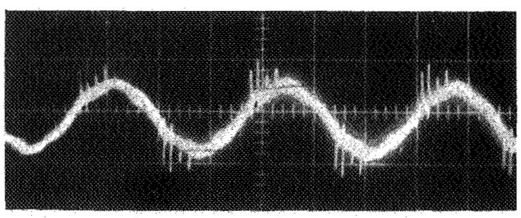
(s) Spacecraft accelerometer; October 15, 1965; 1840 hours G. m. a. t.; 2 milliseconds per centimeter; 50 millivolts per centimeter.



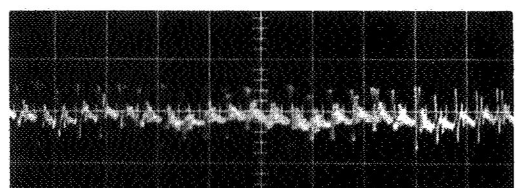
(t) Battery charge sense; October 1, 1965; 1650 hours G. m. a. t.; 1 millisecond per centimeter; 50 millivolts per centimeter (worst case).



(u) Battery charge sense; October 15, 1965; 1835 hours G. m. a. t.; 200 microseconds per centimeter; 50 millivolts per centimeter.



(v) Analog-to-digital converter; October 4, 1965; 1840 hours G. m. a. t.; 5 milliseconds per centimeter; 50 millivolts per centimeter (worst case).



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(w) GSE ground power; October 1, 1965; 1855 hours G. m. a. t.; 1 millisecond per centimeter; 50 millivolts per centimeter.

Figure 19. - Concluded.

on Atlas-Centaur 10 and Surveyor SC-1 at the test facility in San Diego.

Three transients were noted on March 5. These were the result of normal switching of the Surveyor solar panel simulator. Another transient resulted when the programmed command of retro arm on was observed on the operations recorder.

On X - 0 day (Mar. 7), expected transients were noted in response to Surveyor GSE commands of solar panel simulator on/off, retro arm on/off, and Centaur programmer commands of preseparation arming, extend landing gear, and high power transmitter on. However, three transients that were not the result of programmed commands were also observed. Two of these transients occurred on the retro arm on line (-25-V transient threshold), and one occurred on the extend omiantennas line (50-V transient threshold), while the Centaur programmer was in the safe mode.

Inspection of the oscillograph trace of the retro arm on command pulses showed a negative transient at the leading edge of each pulse at the exact time the -25-volt detector monitoring the retro arm on command was triggered.

The 50-volt transient on extend ominantennas occurred during a time interval when Surveyor GSE power cables were being examined to determine if they were securely mated. The Centaur vehicle was passive at this time and could not have generated such a transient.

Random noise of 3.5 volts (peak) was observed on the spacecraft accelerometer line on both days of the test. This noise was traced to a loose connection between the accelerometer transducer and its amplifier. With the amplifier turned off, the noise level fell to 70 millivolts (peak-to-peak). All other steady-state noise levels were below the acceptable levels.

A complete listing of the maximum steady-state and transient interference levels observed during each of the five tests is shown in table IV. The anamolies have been covered in the previous test.

Concurrently with these tests, spacecraft engineers decided that the levels given in table II were neither realistic nor well-defined. Simultaneously, they began a program of further desensitizing their circuits. Much credit must be given them for this effort, for in the end they were so successful that high voltage discharges could be made directly to the spacecraft (fig. 20) without endangering any of their circuits.

Finally, as a result of these tests and also the spacecraft review and desensitizing program, a new set of EMI levels was proposed and agreed to (table III). This table not only distinguishes between steady-state and transient signals, but also identifies signal characteristics as well as source and load impedances. This table was incorporated into the formal JPL's Surveyor Project Control Document No. 1 (Revision 3), which served as the official instrument of agreement between the Surveyor and Centaur projects. The final tests in this series verified electromagnetic compatibility between spacecraft and booster based upon table III.

TABLE IV. - MAXIMUM ALLOWABLE CONDUCTED EMI FROM ATLAS-CENTAUR-GSE ENVIRONMENT TO SURVEYOR[a]

Signal	Signal amplitude level	Signal frequency range	Surveyor allowable steady-state transient EMI levels	Source	Source impedance, Z_S, Ω	Load impedance, Z_L, Ω
Accelerometer output signal	±2.5 V centered about a bias of 2.5±0.1 V dc	5 to 1000 Hz	Steady-state noise fundamental components below 2 kHz shall not exceed 200 mV (peak). No transients shall exceed $^{+100}_{-20}$ V (peak).	Surveyor accelerometer amplifier	200	250×10^3 to 450×10^3
Analog-to-digital converter output signal	Zero state: 0±0.3 V One state: 5±1.0 V	550 pulse/sec	Steady-state noise ±1.5 V (peak). No transients shall exceed ±100 V (peak).	Surveyor central signal processor	1×10^3	250×10^3 to 450×10^3
Centaur commands						
High power transmitter on	26.5±3.5 V	Pulse (100±20 msec)	Transients <100 μsec duration (at 90% amplitude) ±70 V (peak). Transients >100 μsec duration (at 90% amplitude) $^{+18}_{-70}$ V (peak).	Centaur programmer	<1.0	418
Extend landing gear		Pulse (100±20 msec)	Transients <100 μsec duration (at 90% amplitude) $^{+100}_{-50}$ V (peak). Transients >100 μsec duration (at 90% amplitude) ±10 V (peak).			[b]510
Extend omniantennas		Pulse (100±20 msec)	No transient shall exceed $^{+100}_{-50}$ V (peak).			[b]510
Preseparation arming		Pulse (31±0.5 sec)	Transients <100 μsec duration (at 90% amplitude) $^{+60}_{-100}$ V (peak). Transients >100 μsec duration (at 90% amplitude) ±10 V (peak).			[b]510
GSE power						
External battery charge	28 V (maximum)	Direct current	Steady-state noise 2.8 V (peak) below 1.0 kHz and 4.2 V (peak) above 1.0 kHz. No transients shall exceed ±80 V (peak).	GSE ground power supply	<1.0	<1.0
External OCR input	1.7 A, 50 V (average) dc (as constant current source)	Direct current			300	30

Helium vent	40±5 V	Pulse (30 msec)	Transients <100 μsec duration (at 90% amplitude) $^{+60}_{-100}$ V (peak). Transients >100 μsec duration (at 90% amplitude) $^{+30}_{-100}$ V (peak).	GSE helium vent pulse generator	39×10^3	1×10^3 (unactivated) 450 (activated)
Main power switch on/off	28 V	Direct current	Steady-state noise 17.0 V (peak). No transients shall exceed ±100 V (peak).	GSE safety console	2.3×10^3	3.8
Retro igniter safe-and-arm command	28 V	Direct current	Steady-state noise 17.0 V (peak). No transients shall exceed ±50 V (peak) at 50 μsec (or equivalent constant energy.	GSE safety console	∞	28 (closed contact side)
Gyro preheat power	27 V	Direct current	No transients shall exceed ±80 V (peak). Steady-state noise shall not exceed 8.5 V (peak).	GSE systems test equipment assembly	1.0×10^3	40
Battery charge sensing	28 V (maximum)	Direct current	Steady-state noise 2.8 V (peak) below 1.0 kHz. Steady-state noise 4.2 V (peak) above 1.0 kHz. No transients shall exceed ±80 V (peak).	Surveyor battery	<1.0	300×10^3
Retro squib integrity	28 V	Direct current	Steady-state noise 17.0 V (peak). No transients shall exceed ±50 V (peak) at 50 μsec (or equivalent constant energy).	Safe-and-arm-device igniter	<1.0 (squib fired) ∞ (squib unfired)	1×10^3
Safe-and-arm sensing	28 V	Direct current	Steady-state noise 17.0 V (peak). No transients shall exceed ±50 V (peak) at 50 μsec (or equivalent constant energy.	Safe-and-arm device	<1.0 (safe) ∞ (unsafe) <1.0 (arm) ∞ (unarm)	<1.0 (safe) ∞ (unsafe) <1.0 (arm) ∞ (unarm)

[a]Levels listed are to be measured using instrumentation with a minimum bandwidth of 30 mHz.

[b]Z_L = 250±50 Ω when extend landing gear and preseparation arming or extend ominatennas and preseparation landing are on simultaneously.

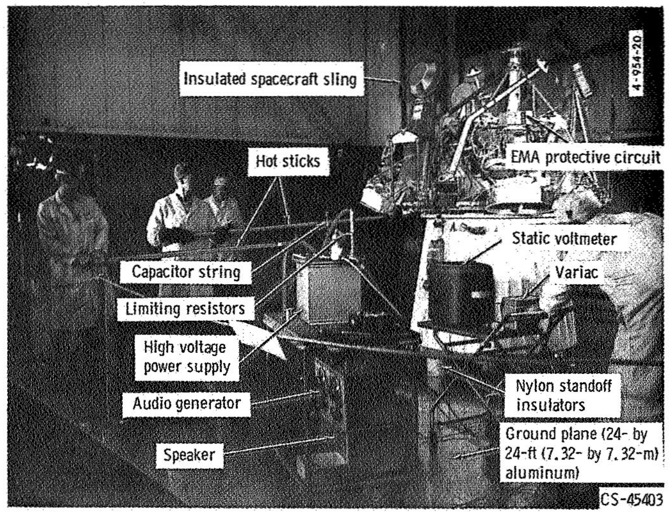

Figure 20. - HAC electrostatic test setup for Surveyor spacecraft.

CONCLUSIONS

A series of five instrumented electromagnetic compatibility tests were conducted on interface circuits between the Surveyor spacecraft and Atlas-Centaur launch vehicle. These tests were performed during ground checkout procedures at the launch complex at Cape Kennedy and the test facility in San Diego, California. Initial tests employed spacecraft and launch vehicle simulators with their complementary vehicle, while the final test was performed on flight equipment.

The major conclusions of this test program are the following:

1. Both steady-state and transient signals were examined and found to be less than one-half of the spacecraft susceptibility levels. Compatibility was demonstrated in that no adverse signals were found. No corrective action was required for Atlas, Centaur, or associated Ground Support Equipment (GSE). (Surveyor modifcations were made.)

2. These tests established that both Centaur and Surveyor were designed in accordance with sound electromagnetic capability (EMC) practice, and even though designed independently, were, in fact, compatible (after Surveyor modification).

3. It was learned that arbitrary test levels may not be realistic, and that practical EMC specifications are useful only if they can be verified by test.

4. All participants recognized that EMC depends upon electromagnetic coordination. Such coordination and cooperation was obtained in this program.

5. It is important to allow enough time in major multisystem programs such as Atlas-Centaur-Surveyor for thorough EMC testing and possible redesign and retesting.

6. For maximum confidence in test results, the degree of simulation should be minimized, and actual final configurations employed wherever possible.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, June 19, 1968,
491-03-00-01-22.

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